UNCLASSIFIED

AD NUMBER

ADB002859

NEW LIMITATION CHANGE

TO

Approved for public release, distribution unlimited

FROM

Distribution authorized to U.S. Gov't. agencies only; Test and Evaluation; OCT 1974. Other requests shall be referred to Air Force Materials Laboratory, Attn: LC, Wright-Patterson AFB, OH 45433.

AUTHORITY

AFML, DoDD 5200.20

THIS PAGE IS UNCLASSIFIED

AFML-TR-74-164

CONCEPTUAL DESIGN STUDIES OF COMPOSITE AMST

MCDONNELL DOUGLAS CORPORATION
DOUGLAS AIRCRAFT COMPANY
LONG BEACH, CALIFORNIA 90846

OCTOBER 1974

TECHNICAL REPORT

Distribution limited to U.S.Government agencies; test and evaluation; statement applied June 1974. Request for this document must be referred to AFML (LC), Wright-Patterson Air Force Base, Ohio 45433.

AIR FORCE MATERIALS LABORATORY
AIR FORCE SYSTEMS COMMAND
WRIGHT—PATTERSON AIR FORCE BASE, OHIO 45433

This report has been reviewed and is approved.

David A. Roselius
Project Engineer
Advanced Development Division
Air Force Materials Laboratory

Major Richard A. Mollicone, USAF
Program Manager
Advanced Development Division
Air Force Materials Laboratory

NOTICES

AIR FORCE/56780/24 March 1975 — 100

UNCLASSIFIED
SECURITY CLASSIFICATION OF THIS PAGE (When Data Entered)

REPORT DOCUMENTATION PAGE		READ INSTRUCTIONS BEFORE COMPLETING FORM
1. REPORT NUMBER AFML-TR-74-164	2. GOVT ACCESSION NO.	3. RECIPIENT'S CATALOG NUMBER
4. TITLE (and Subtitle) CONCEPTUAL DESIGN STUDIES OF COMPOSITE AMST		5. TYPE OF REPORT & PERIOD COVERED Final Technical Report of Work Performed between 29 May 1973 and 23 May 1974
		6. PERFORMING ORG. REPORT NUMBER MDC J-4446
7. AUTHOR(s) NELSON, W. D., et al		8. CONTRACT OR GRANT NUMBER(s) F33615-73-C-5164
9. PERFORMING ORGANIZATION NAME AND ADDRESS McDonnell Douglas Corporation Douglas Aircraft Company Long Beach, California 90846		10. PROGRAM ELEMENT, PROJECT, TASK AREA & WORK UNIT NUMBERS
11. CONTROLLING OFFICE NAME AND ADDRESS Air Force Materials Laboratory Air Force Systems Command Wright-Patterson Air Force Base, Ohio 45433		12. REPORT DATE OCT 1974
		13. NUMBER OF PAGES 203
14. MONITORING AGENCY NAME & ADDRESS (if different from Controlling Office)		15. SECURITY CLASS. (of this report) UNCLASSIFIED
		15a. DECLASSIFICATION/DOWNGRADING SCHEDULE

16. DISTRIBUTION STATEMENT (of this Report)

Distribution limited to U.S. Government agencies; test and evaluation; statement applied June 1974. Request for this document must be referred to AFML (LC), Wright-Patterson Air Force Base, Ohio 45433.

17. DISTRIBUTION STATEMENT (of the abstract entered in Block 20, if different from Report)

18. SUPPLEMENTARY NOTES

19. KEY WORDS (Continue on reverse side if necessary and identify by block number)

Advanced Composites	Graphite-Epoxy	Performance Payoffs
Low-Cost Manufacturing	Detailed Cost Estimates	Maintenance Man-hours Per Flight Hour
Conceptual Design	Life Cycle Costs	
C-15 (AMST) Aircraft	Acquisition Costs	
	Aircraft Resizing	

20. ABSTRACT (Continue on reverse side if necessary and identify by block number)

A CURRENT DOUGLAS AMST PRODUCTION AIRCRAFT CONFIGURATION WAS USED AS A BASELINE TO DETERMINE VEHICLE PERFORMANCE AND COST IMPROVEMENTS ACCRUING FROM THE MAXIMAL USE OF ADVANCED COMPOSITE MATERIALS IN THE AIRFRAME THE PRIMARY WING AND EMPENNAGE BOX STRUCTURE AND FUSELAGE SHELL APPLICATIONS WERE EMPHASIZED TOGETHER WITH SELECTED APPLICATIONS IN SECONDARY STRUCTURES TO REDUCE THE WEIGHT OF THE AIRFRAME THE PROPERTIES OF HIGH-STRENGTH GRAPHITE-EPOXY COMPOSITES (REPRESENTATIVE OF THORNEL 300 FIBERS) WERE USED IN THE APPLICATION STUDIES. MATERIAL COSTS REPRESENTATIVE OF THORNEL 300/EPOXY PREPREG AND A LOWER COST PITCH-BASED FIBER/EPOXY PREPREG WERE USED IN THE COST ANALYSES. COST REDUCTION WAS EMPHASIZED IN THE COMPOSITE STRUCTURAL DESIGN SELECTIONS AND THE

DD FORM 1 JAN 73 1473 EDITION OF 1 NOV 65 IS OBSOLETE
S/N 0102-014-6601

UNCLASSIFIED
SECURITY CLASSIFICATION OF THIS PAGE (When Data Entered)

INNOVATIVE MANUFACTURING TECHNIQUES REFLECTED THAT EMPHASIS. THE WING AND HORIZONTAL STABILIZER BOX STRUCTURES WERE TRUSS-WEB CONCEPTS, THE VERTICAL STABILIZER BOX STRUCTURE WAS A SANDWICH-PANEL MULTICELL DESIGN (ASSEMBLED THROUGH THE USE OF INTERNAL PRESSURE BAGS), AND THE FUSELAGE SHELL WAS AN AUTOMATICALLY TAPE-WRAPPED ISOGRID CONCEPT. THREE COMPOSITE AIRCRAFT CONFIGURATIONS WERE DEFINED ONE WITH EXTERNAL DIMENSIONS IDENTICAL TO THE BASELINE, A SECOND ONE REDUCED IN SIZE FOR FULL EXPLOITATION OF THE WEIGHT SAVING (INCLUDING REDUCED SCALE ENGINES), AND A THIRD ONE PARTIALLY RESIZED BASED ON EXISTING BASELINE ENGINES MANUFACTURING COST ESTIMATING DRAWINGS WERE PREPARED IN SUFFICIENT DETAIL TO DEFINE SIGNIFICANT STEPS REQUIRED IN TOOLING AND MANUFACTURING. DETAILED WEIGHT AND COST ESTIMATES WERE PREPARED FOR THE BASELINE CONFIGURATION, THE THREE COMPOSITE AIRCRAFT CONFIGURATIONS USING THORNEL 300/EPOXY COMPOSITES, AND THE FULLY RESIZED COMPOSITE CONFIGURATIONS USING PITCH-BASED FIBER/EPOXY COMPOSITES. THE UNRESIZED COMPOSITE AIRCRAFT INDICATED A 10.7-PERCENT REDUCTION IN STRUCTURAL WEIGHT BUT COST INCREASES OF 2.4 AND 0.9 PERCENT, RESPECTIVELY, ON UNIT AND LIFE-CYCLE COSTS FOR 300 AIRCRAFT OPERATING FOR A 20-YEAR PERIOD. THE WEIGHT REDUCTION CONVERTED INTO A 6-PERCENT REDUCTION IN TAKEOFF LENGTH, A 20.4-PERCENT INCREASE IN PAYLOAD, OR A 46.3-PERCENT INCREASE IN MISSION RANGE. THE FULLY RESIZED COMPOSITE AIRCRAFT USING PITCH-BASED FIBERS MET THE BASIC MISSION PERFORMANCE WITH A 14.8-PERCENT REDUCTION IN STRUCTURAL WEIGHT, A UNIT-COST REDUCTION OF 7.5 PERCENT, AND A LIFE-CYCLE COST REDUCTION OF 4.9 PERCENT ($474 MILLION). THE RESIZED AIRCRAFT USING THORNEL 300 FIBERS AND BASELINE ENGINES INDICATED THE GREATEST STRUCTURAL WEIGHT SAVINGS (15.4 PERCENT) BUT UNIT AND LIFE-CYCLE COSTS ONLY SLIGHTLY LESS THAN THE BASELINE METAL CONFIGURATION.

PREFACE

This Technical Report was prepared by Douglas Aircraft Company, McDonnell Douglas Corporation, Long Beach, California, under contract No. F33615-73-C-5164, for the Advanced Development Division, Air Force Materials Laboratory, Air Force Systems Command, Wright-Patterson Air Force Base, Ohio. Mr. D. A. Roselius (AFML/LC) was the Air Force Project Engineer and Mr. W. D. Nelson was the Douglas Aircraft Company Program Technical Manager.

Work performance spanned the period 29 May 1973 to 23 May 1974.

Principal contributors to the Douglas activities described in this report were:

Structural Design	W. D. Nelson, H. W. Wilson
Structural Mechanics	Dr. L. J. Hart-Smith, A. Cominsky
Weights Engineering	P. W. Scott
Economic Analysis	M. M. Platte, W. O. Welly
Aerodynamics/Configuration	J. H. Lindley
Advanced Design/Configuration	W. D. Kelly
Materials and Process Engineering	R. J. Palmer
Manufacturing Research and Development	A. T. Tucci
Advanced Planning	R. A. Dobbs, R. E. Kent
Manufacturing Estimating and Cost Evaluation	G. E. Frazee, R. O. Lines
Industrial Engineering Tooling and Planning	R. D. Klein, G. A. Hawk
Quality and Reliability Assurance	E. G. Holden

SUMMARY

The primary objectives of the program were (1) to develop advanced composite primary airframe structural concepts offering vehicle performance improvements, increased reliability, and reduced cost, and (2) to develop structural design concepts from the standpoint of manufacturing cost reduction. A current Douglas Advanced Medium STOL Transport (AMST) production aircraft configuration was used as a baseline to determine vehicle performance and cost improvements accruing from the maximal use of advanced composite materials in the airframe.

The work was conducted in four tasks as follows:

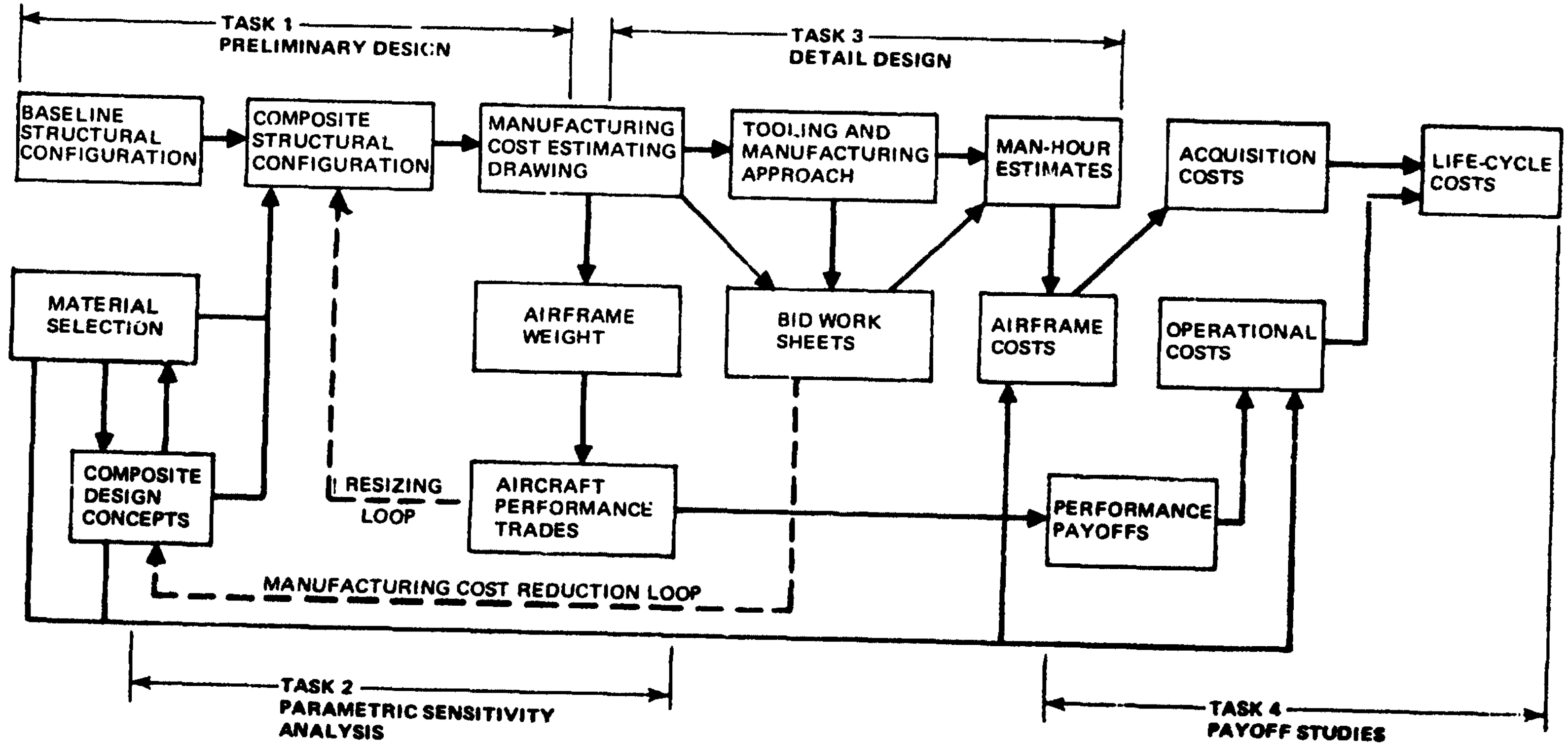
- Task I, Preliminary Design, documented the baseline metal aircraft configuration, established composite primary structure design and manufacturing concepts, and formulated an initial composite aircraft configuration.
- Task II, Parametric Sensitivity Analysis, optimized or improved the performance or geometry of the initial composite aircraft configuration. The performance effects of perturbing aerodynamic parameters were determined.
- Task III, Detail Design, detailed the composite design tooling, and manufacturing concepts to provide bases for weight and cost estimates.
- Task IV, Payoff Studies, established composite aircraft payoffs in performance and cost with respect to the baseline metal aircraft.

The work progression through the various tasks is illustrated in the accompanying program flow diagram.

Geometry, weights, and engineering data from a current AMST production configuration were utilized to establish a baseline metal airplane. An initial composite airplane, dimensionally equal to the baseline, was also defined to provide geometry, load, and weight bases for the composite design studies, structural analyses, and cost estimating. The latter airplane was initially resized to meet the basic airplane mission on the assumption of a 12-percent reduction in manufacturers empty weight. This assumption was later verified by the final program results.

The primary advanced composite material selected was a high-strength graphite-epoxy (Thornel 300/Narmco 5208). Boron-infiltrated aluminum extrusions were specified in the cargo floor and loading ramp, and conventional metallic materials were retained in many areas. Overall utilization of composites amounted to 42 percent of the structural weight in the final configuration. Low-cost pitch-based graphite fiber was also considered, together with low-cost glass/graphite and Kevlar/graphite hybrids to provide a second cost category for the composite aircraft.

Utilizing a cost selection rationale based on industrial engineering cost estimating techniques, composite design concepts were selected for the primary



PROGRAM FLOW DIAGRAM

wing and empennage box structures and the fuselage shell. The selected design concepts were truss-web wing and horizontal tail box structures, a multirib/multispar vertical tail box structure, and a wrapped isogrid fuselage shell. Various secondary structures were converted to composite on a detail design basis, including integrally molded multirib control surfaces. The selected design concepts for the primary structures are illustrated in the accomparying figures. The low cost potential of the integral derby-hat-stiffened laminate panel was highlighted in this study. Derby-hat-stiffened panels were lower in cost than honeycomb panels, if the honeycomb panel required machine tapering, utilized nonmetallic core, or had edge treatment and potted insert areas.

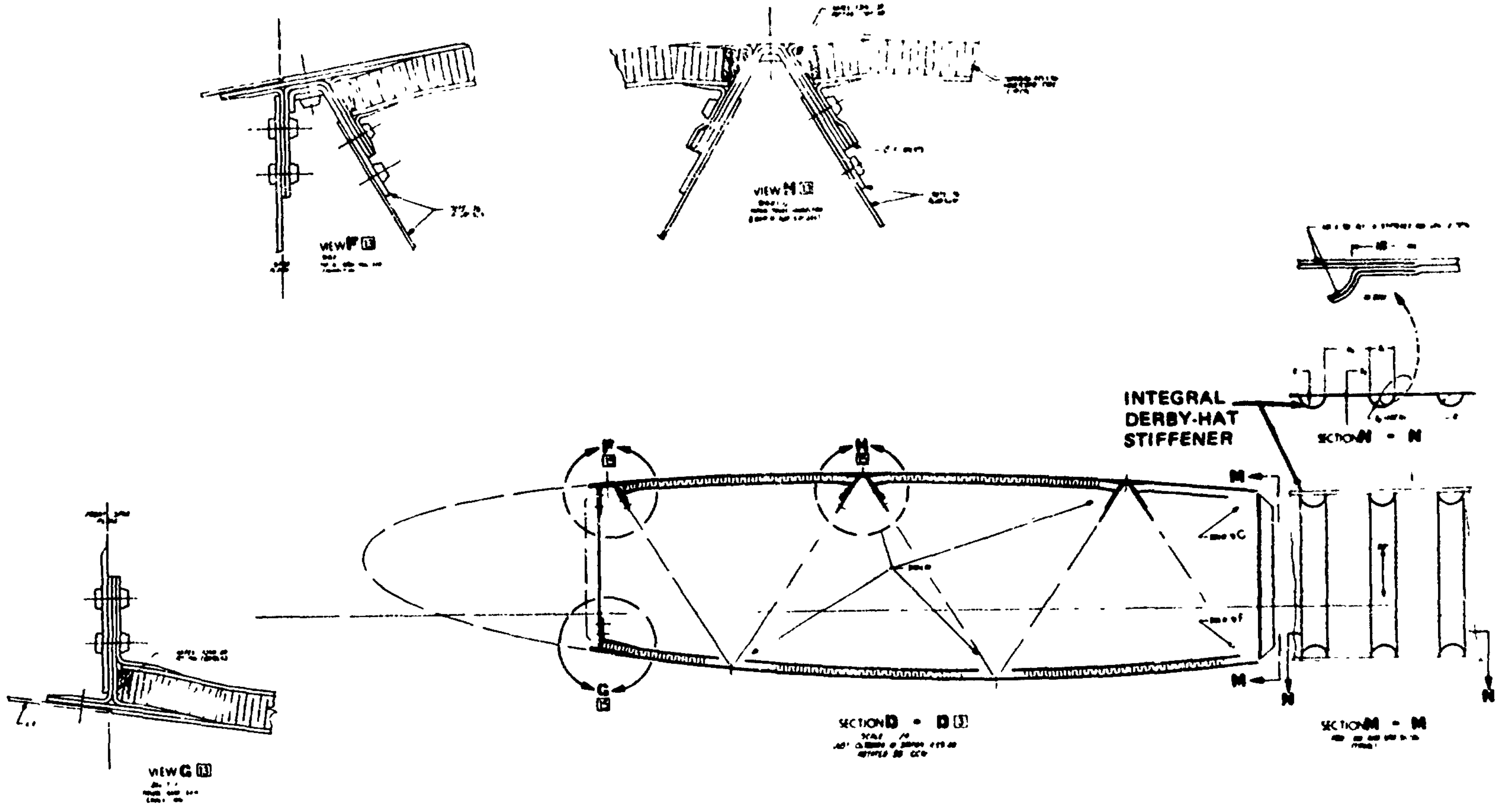
Aerodynamic configuration sensitivity analysis showed wing and empenage size (but not geometry) to be affected by composite applications. The fuselage was not resized (in spite of the frameless isogrid design) because of cargo loading requirements and because wing/fuselage/cargo box geometry interactions adversely affected weight and drag.

With the selection cf composite design concepts, manufacturing cost estimating (MCE) drawings were created. The drawings delineated essential structural details which impact manufacturing costs (e.g., patterns, thicknesses, and joints). The cost estimating methodology utilized an industrial engineering approach to generate basic "bottoms-up" estimates of composite component tooling and manufacturing costs.

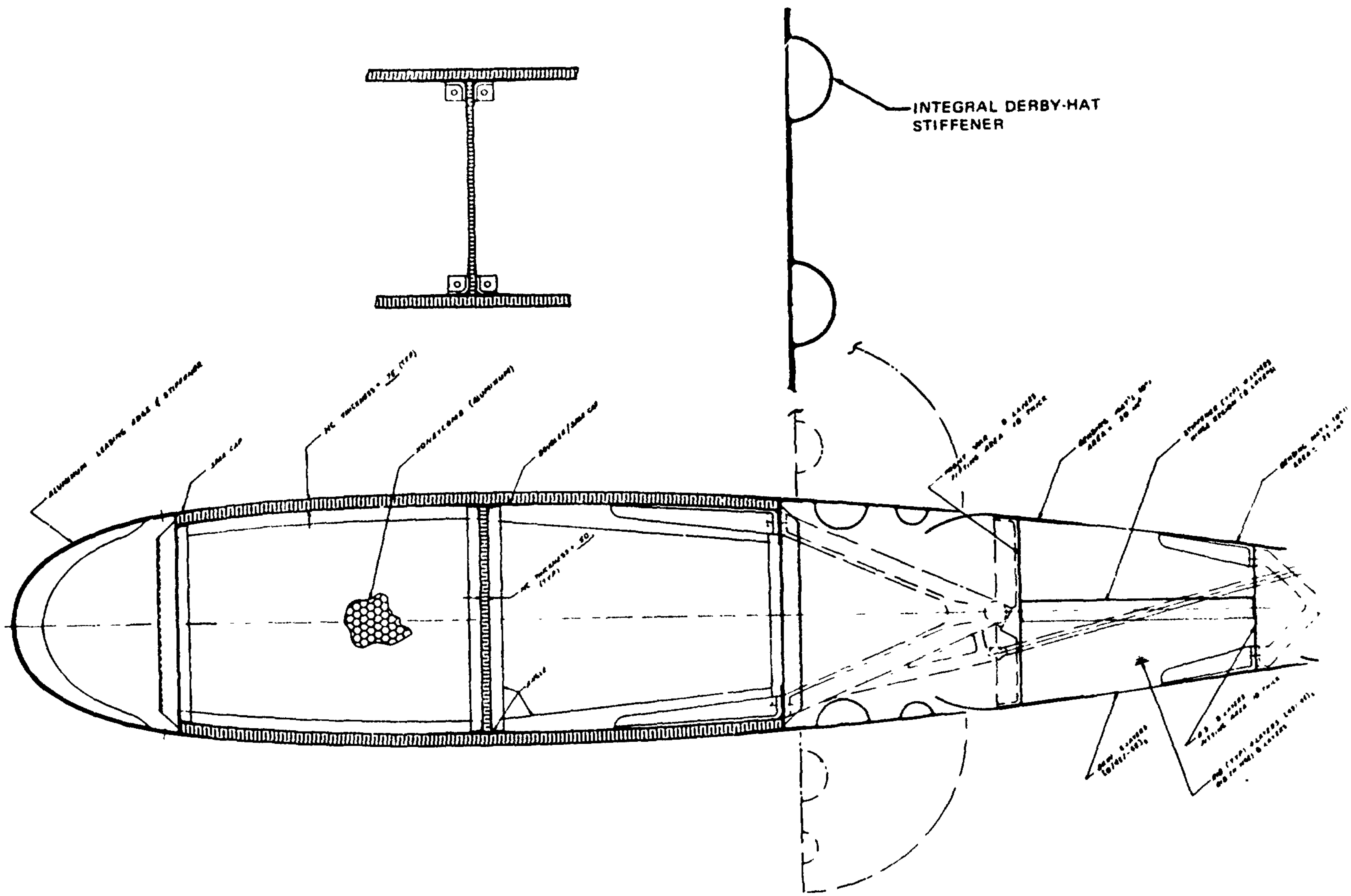
The manufacturing concepts included electrical discharge machining (EDM) of large honeycomb panels, automatic tape layup of foot-wide broadgoods, ply thickness tailoring to the component size and thickness, and pultrusion of cross-plied laminate patterns in both constant and tapered sections. Aluminum alloy construction was retained where warranted by lower cost and design function, such as in the truss-web box substructure.

An inflatable internal bag concept for cocuring and bonding the vertical tail box structure was introduced to reduce recurring assembly costs. A fuselage shell manufacturing labor reduction of approximately 31 percent was indicated by use of the automated winding system shown in the accompanying figure. Large integrally stiffened structures were emphasized throughout the composite design to reduce the parts count and the number of joints.

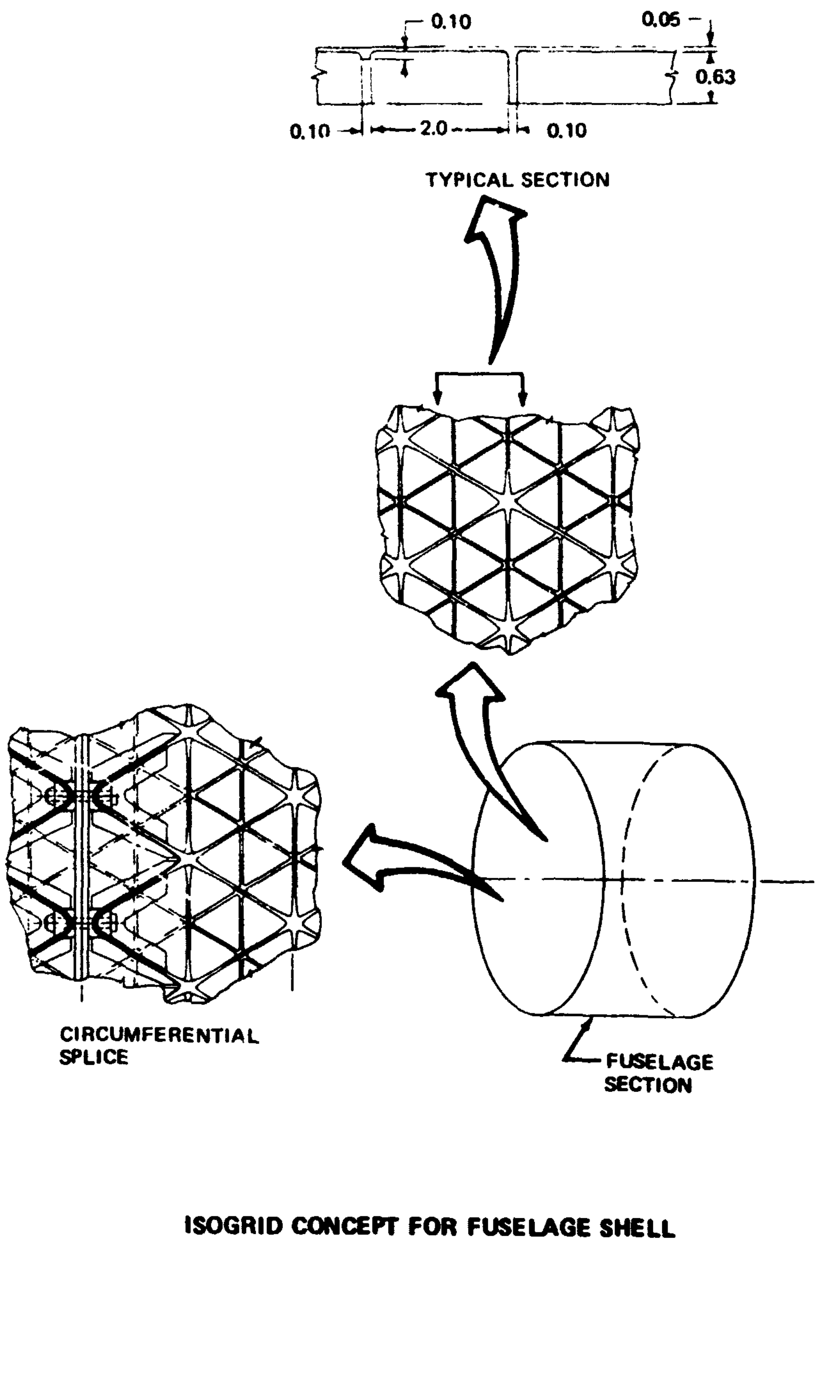
Structural analysis was based on a detailed finite element analysis conducted on the metal YC-15 prototype. Loads and element sizes were scaled appropriately for the initial resized composite aircraft. The analysis validated the MCE drawings and guided the weights calculations. Stiffness, rather than strength, was found to be critical in wing and tail box structure sizing. Material was subsequently removed from the initial stiffness designs of the primary box structures since the flutter margins of the composite designs were overly conservative. Adequate flutter margins were attained when approximately 40-percent thickness was removed from the initially sized wing skins.



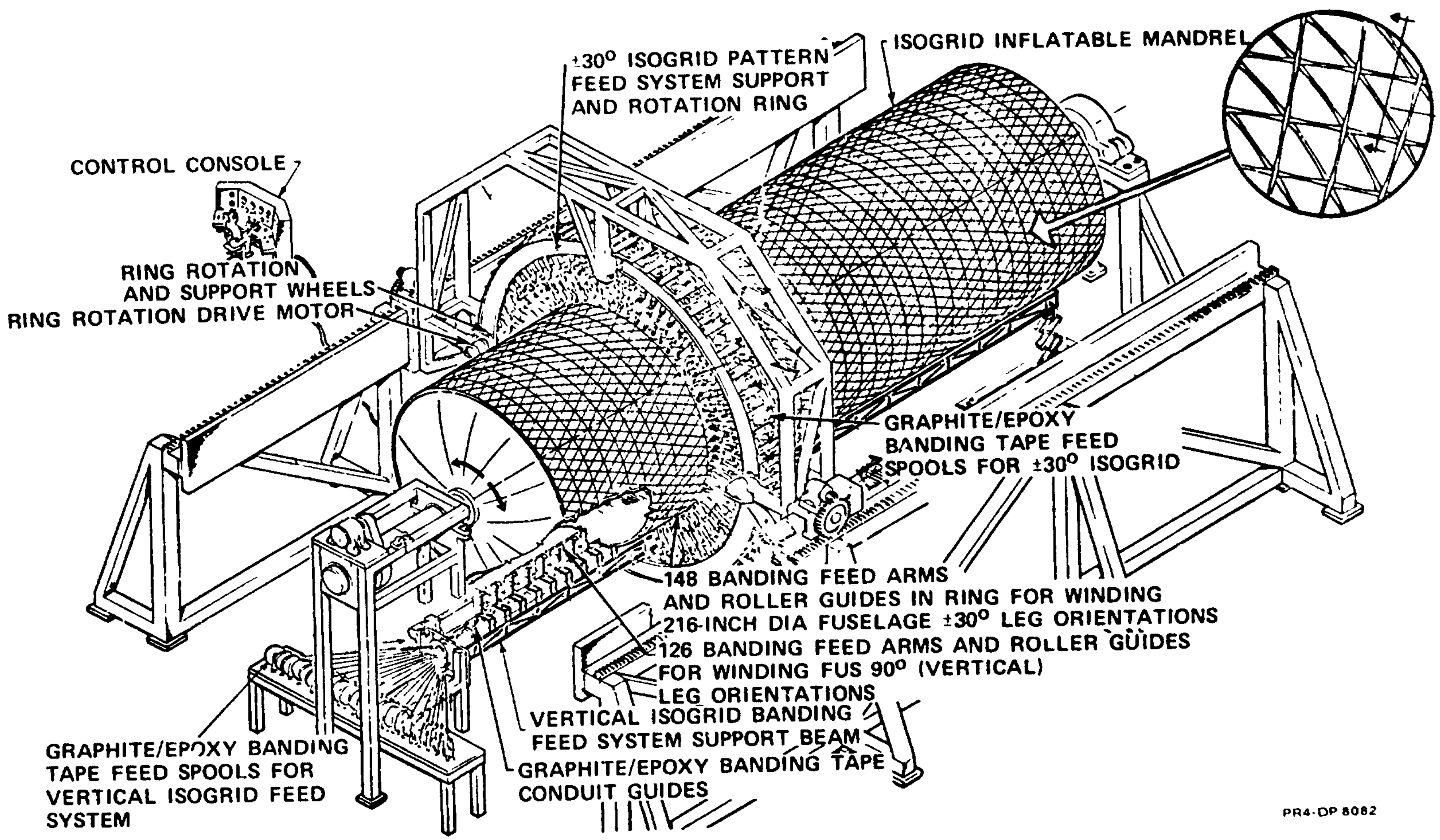
STRUCTURAL CONCEPT FOR WING AND HORIZONTAL STABILIZER BOX STRUCTURES



STRUCTURAL CONCEPT FOR VERTICAL STABILIZER BOX-STRUCTURE



ISOGRID CONCEPT FOR FUSELAGE SHELL



ISOGRID FUSELAGE AUTOMATED WINDING CONCEPT

Most of the composite isogrid fuselage design was governed by stability or strength considerations. Manufacturing cost considerations required that the grid dimensions be maintained constant over large areas of the shell. Consequently some regions of the fuselage were somewhat overstrength (and over weight) because of the preference for manufacturing cost reduction.

A detailed weight statement was prepared for the initial composite aircraft configuration based on the structural details of the MCE drawings. From this information and the performance tradeoff studies, two additional composite aircraft configurations were formulated based on equal STOL mission performance. The resulting three composite aircraft were:

- A configuration dimensionally equivalent to the baseline aircraft (unresized) using the existing JT8D-17 engines.

- A fully resized configuration for full conversion of the weight saving into size reduction using hypothetical reduced-scale engines.

- A partially resized configuration optimized around the existing JT8D-17 engines.

For the unresized composite aircraft, a structural weight reduction of 5760 pounds was realized. The performance tradeoff studies indicated that this weight reduction may be converted into a reduction in the required field length (1880 feet rather than 2000), an increase in payload (32,560 pounds rather than 27,000), or an increase of basic mission radius (585 nautical miles rather than 400).

The fully resized aircraft met the basic mission performance with a wing area of 1607 square feet (rather than 1740) and reduced size engines (scaled-down JT8D-17's). This configuration indicated a 7.7-percent reduction in STOL takeoff gross weight, a 10.4-percent reduction in manufacturer's empty weight (MEW), a major component weight savings of 9.9 percent in the fuselage, 20.5 percent in the wing, 20.7 percent in the vertical tail, and 21.8 percent in the horizontal tail. Higher percentage savings were realized in the primary box structures and the control surfaces.

In comparison to the fully resized aircraft, the resized aircraft with existing JT8D-17 engines showed slightly reduced savings in takeoff weight, MEW, and fuselage weight, but increased weight savings in the wing and empennage structures. The larger engine permitted an increase in wing loading (90.5 pounds per square foot rather than 86.0) with a consequent reduction in wing area to 1545 square feet. Composite utilization ranged from 73.3 percent in the vertical tail box structure to 30.1 percent in the cargo floor and supports, averaging 42 percent of total structural weight.

A cost analysis was conducted to compare estimated costs for development, accuisition, and operations of the various AMST study configurations and to substantiate the economic feasibility of extensive use of advanced composite materials in the airframe.

Five aircraft configurations were considered in the cost analysis: the baseline metal configuration, the three composite configurations described above utilizing Thornel 300/Narmco 5208 graphite-epoxy material, and the fully resized composite configuration using a pitch-based (lower cost) graphite fiber. The unit price for the baseline metal airplane was established using the costing/methodology formulated for this program.

The configuration characteristics of the development, procurement, operations, and support costs for the five configurations are summarized in the accompanying table. Although significant weight savings (up to 40 percent) were realized in some components, these savings did not yield an equal reduction in system weights. The aircraft manufacturer's planning report (AMPR) weight of the resized aircraft was 12 percent less than the baseline. The operating empty weight was 10.4 percent less and takeoff gross weight (TOGW) was only 7.7 percent less than the baseline aircraft. The avionics, engines, systems, and nonstructural portions of the aircraft weight tended to dilute the impact of the structural weight saving.

The reverse trend was indicated in the cost analysis. Although the material costs of the advanced composite aircraft were considerably higher than the baseline metals, the structural material cost impact was diluted by the effect of the avionics, engines, aircraft systems, and other nonstructural portions of the aircraft. Thus, the total unit price of the aircraft (even with the more expensive Thornel 300 material) was the same or lower than the unit price of the baseline metal airplane. The lower priced pitch-based composite material yielded an acquisition cost 6.9 percent less than the cost of the baseline metal airplane.

System operational costs were projected using the Air Force PACE (Planning Aircraft Cost Estimating) model using a conservative estimate of maintenance man-hours per flying hour. The composite airframe maintenance man-hours showed increases of 36 to 38 percent across the configurations. Total maintenance functions increased 6 to 9 percent.

Before the economic benefits of advanced composite technology can be realized, substantial utilization of composite materials must be effected. In particular, advanced composite must be used in the primary structure to permit resizing of the entire vehicle. Unless the airplane was resized to take advantage of the composite material properties, increase in total system cost resulted. Out of a total baseline cost of $9.66 billion for 20 years operations, approximately $197.7 million were saved by using the Thornel 300 material with a 30- to 50-percent composite utilization. The less expensive pitch-base fiber material (or alternatively, an equivalent drop in Thornel 300 prices) produced additional savings on the order of $276 million over the life-cycle of the system. Still higher aircraft operating savings are available if current escalating fuel costs are considered.

Cost estimates for the metal baseline and the advanced composite aircraft in this study are based on historical data and detailed discrete component estimates for the airframe and the airframe systems, engine company prices for the propulsion system, and subcontractor cost data for avionics. This provides a consistent and solid approach for developing and comparing the differential weights and costs between these aircraft.

The metal baseline aircraft is similar in physical characteristics and performance to the projected C-15 production aircraft; however, it should be recognized that since the inception of this study the C-15 aircraft has been tailored to a production "design-to-cost" program. This program emphasizes primary cost reductions compared to traditional design and program concepts and therefore $10.1M in FY 1973 dollars for the study baseline aircraft compares to a $6.6M price in FY 1972 dollars for the C-15 "design-to-cost" production aircraft.

For purposes of this study the specific intent was to establish and maintain throughout a consistent base for comparing the metal baseline and advanced composite aircraft wherein the results obtained are indicative of the potential for composite application. However, beyond that, it is essential in the future to examine the potential of adapting composites to a projected design-to-cost aircraft by recognizing the challenge to achieve a 25-percent reduction in airframe cost commensurate with a design-to-cost program. A final proof of achievement would entail design, fabrication, test, evaluation and cost tracking of full-scale primary structural components with "design- and produce-to-cost" a primary parameter.

CONFIGURATION CHARACTERISTICS AND COST SUMMARY

	CONFIGURATION				
	METAL BASELINE	UNRESIZED ADVANCED COMPOSITE THORNEL 300	RESIZED ADVANCED COMPOSITE		PARTIALLY RESIZED WITH FIXED ENGINE THORNEL 300
			THORNEL 300	PITCH-BASED FIBER	
THRUST/ENGINE – SLS, LB	14,900	14,900	13,760	13,760	14,900
WEIGHT SUMMARY – LB					
AMPR WEIGHT	79,016	73,269	70,064	70,064	70,033
MFG WEIGHT EMPTY	98,726	92,977	88,487	88,487	89,515
OPERATOR'S WEIGHT EMPTY	103,2[illegible]	97,487	92,980	92,980	94,000
TAKEOFF GROSS WEIGHT	150,000	150,000	138,500	138,500	139,890
COST WEIGHT	82,055	76,309	73,095	73,095	73,072
COST SUMMARY – JAN 1, 1973 DOLLARS					
RDT&E (5 AIRCRAFT)	$ 657.583 M	$ 672.105 M	$ 663.793 M	$ 649.860 M	$ 665.766 M
PRODUCTION (295 AIRCRAFT)	3550.984	3719.650	3576.383	3367.160	3604.524
ACQUISITION SUBTOTAL	$4208.567 M	$4391.755 M	$4240.176 M	$4017.020 M	$4270.290 M
OPERATIONS AND SUPPORT (20 YR)	$5132.446 M	$5193.431 M	$5066.078 M	$5014.004 M	$5177.201 M
TOTAL LIFE CYCLE	$9341.013 M	$9585.186 M	$9306.254 M	$9031.024 M	$9447.491 M
PRODUCTION UNIT PRICE	$10.090 M	$10.601 M	$10.187 M	$9.554 M	$10.252 M

CONTENTS

LIST OF ILLUSTRATIONS

ILLUSTRATIONS (Continued)

ILLUSTRATIONS (Continued)

ILLUSTRATIONS (Continued)

LIST OF TABLES

LIST OF TABLES (Continued)

LIST OF TABLES (Continued)

LIST OF TABLES (Continued)

SECTION 1

INTRODUCTION

This program had two objectives:

1. To develop advanced composite primary airframe structure designs and concepts offering high payoff in terms of vehicle performance improvements, increased reliability, and reduced cost.

2. To develop airframe concepts from the standpoint of manufacturing cost reduction.

The baseline airframe for this effort was a production configuration of a Douglas Advanced Medium STOL Transport (AMST).

The four tasks under which the work was organized were identified as follows:

Task I - Preliminary Design. This task consisted of documenting the baseline metal aircraft configuration and establishing composite primary structure design and manufacturing concepts. An initial composite AMST study airplane, sized to perform the basic mission, was also established. These aircraft configurations and the composite material and design concept selections are discussed in Section 2, Preliminary Design.

Task II - Parametric Sensitivity Analysis. This task was conducted to optimize or improve the performance or geometry of the initial composite airplane and to optimize the aircraft for maximum cost reduction. The effects of perturbing aerodynamic parameters of the basic wing and fuselage are summarized in Section 3, Configuration Parametric Sensitivity Analysis.

Task III - Detail Design. This task detailed the composite concepts in sufficient depth to provide a basis for final weight and cost estimates. A structural analysis was conducted, manufacturing approaches were defined, and cost estimates were prepared. The results of these studies are presented in Section 4, Detail Design.

Task IV - Payoff Studies. This task established payoffs in performance and cost of the composite aircraft with respect to the baseline metal aircraft. The baseline metal aircraft and three composite configurations varying in overall size were evaluated. One composite configuration was evaluated using both current and projected (reduced) costs for the raw materials. The performance and cost comparisons are presented in Section 5, Payoff Studies, and Section 6, Cost Analysis.

Although this was a conceptual design program for composite structures, cost reduction was twice emphasized in the objectives, once from the component initial manufacturing cost standpoint and again from a total vehicle system basis. Accordingly, an aggressive design/manufacturing interface was established to guide the cost selection of design and manufacturing concepts, and the resulting concept selections for the primary wing and empennage box structures and fuselage shell reflect this approach.

SECTION 2
PRELIMINARY DESIGN

2.1 BASELINE AIRPLANE

Figure 1 is the general arrangement of the baseline metallic AMST aircraft and Figure 2 shows a cutaway drawing. The configuration is characterized by a high wing, four JT8D-17 engines, and a T-tail, and features a large cross section fuselage, rear-end cargo loading, and high flotation landing gears. A supercritical wing airfoil section is utilized to provide reasonable cruise speeds and sufficient fuel volume for the ferry mission. The wing and horizontal stabilizer have straight leading and trailing edges and rear spars which are normal to the aircraft centerline. The vertical stabilizer is a constant-chord, constant-thickness surface.

The cargo box is 560 inches long and is contained in a 216-inch-diameter fuselage. This baseline metal airplane configuration is similar to a previous "design to cost" study airplane. This analysis and data from the YC-15 prototype effort conducted concurrently also benefited the presently reported program.

2.2 INITIAL COMPOSITE AIRPLANE

A preliminary composite aircraft configuration was required to scale the loads from the metallic baseline airplane. Aircraft resizing was performed based on a 12-percent reduction in manufacturer's empty weight, while maintaining the same field length and design mission performance as the metallic aircraft. This 12-percent weight reduction was selected based on experience gained during two previous composite aircraft studies (References 1 and 2) and did not represent the component weights yet to be derived from Task III detail design. The initial resized aircraft has an 8-1/3-percent lower takeoff gross weight, wing area, and engine size than the metal baseline as shown in Table 1. Figure 3 is a general three-view of the initial composite airplane. This preliminary airplane formed the basis for the structural design and analysis effort.

2.3 MATERIAL SELECTION

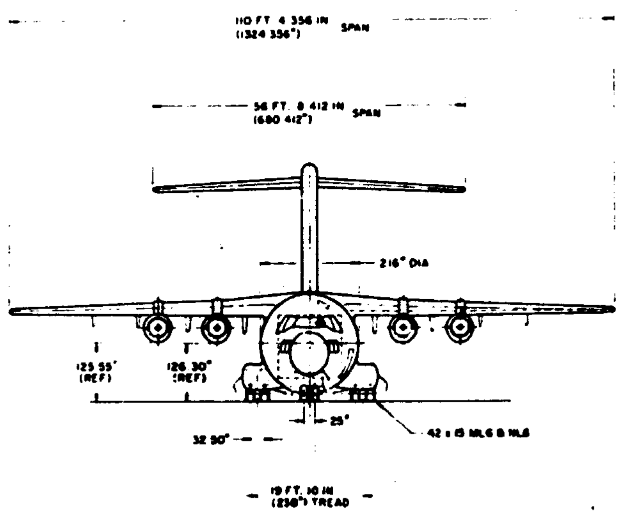
High-strength graphite-epoxy (the Thornel 300/Narmco 5208 system) was the primary composite material selected for use in this program based on previous IRAD material evaluation work and its subsequent qualification at Douglas for another program. Primary considerations in the choice of T300/5208 were the excellent mechanical properties, low cost, and shop-readiness accompanied by current production of significant quantities. Additional considerations were the good material handling characteristics, processing characteristics, and miscellaneous factors such as availability of data from complementary IRAD programs, Douglas experience with the product, and the history of supplier service and corporate backing.

Four-mil boron-epoxy was considered the high modulus material selection, either for use by itself or in hybrid laminate mixtures with graphite, if select areas for cost-effective application developed; however, the only generally apparent area that developed for boron filament application was as infiltrated reinforcement of aluminum extrusions for the cargo floor and

ITEM	WING (BASIC)	HORIZONTAL TAIL	VERTICAL TAIL
AREA SQ FT	1740	643	462
ASPECT RATIO	7.0	5.0	0.894
TAPER RATIO	0.3	0.45	1.0
SWEEP ¼	5° 53' 6"	4° 46' 12"	41°
ANHEDRAL	0°	3°	~
THICKNESS % CHORD	13.908% AVERAGE	11%	12%
INCIDENCE	11 9/2 = 3.437° 323 9/2 = 4.147° 95 9/2 = 1.311°	+5° TO -15°	~
VOLUME RATIO	~	1.323	.1235

CHARACTERISTICS DATA

CARGO COMPARTMENT SIZE
540" LENGTH (EXCLUDES WALKWAY)
140" WIDTH
136" HEIGHT (MIN)
146" HEIGHT (MAX)



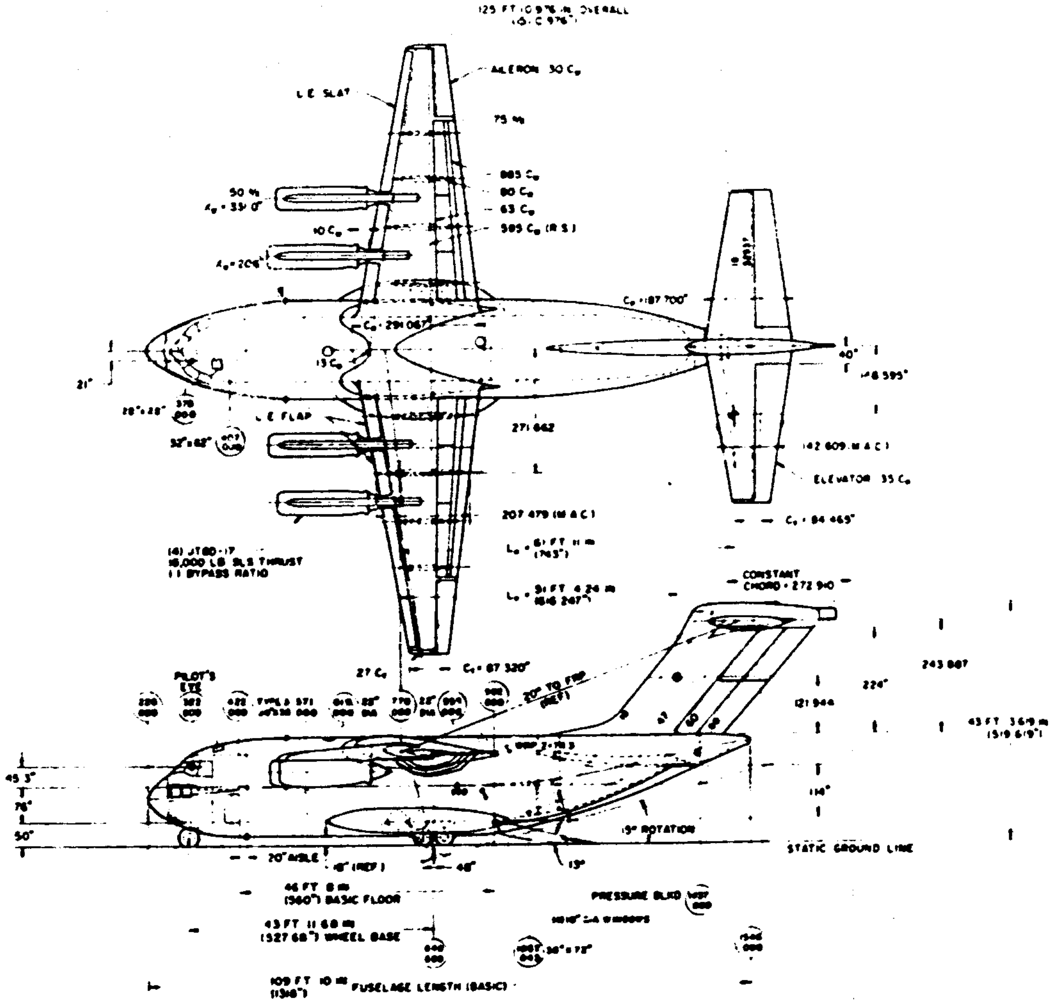


FIGURE 1. GENERAL ARRANGEMENT – AMST

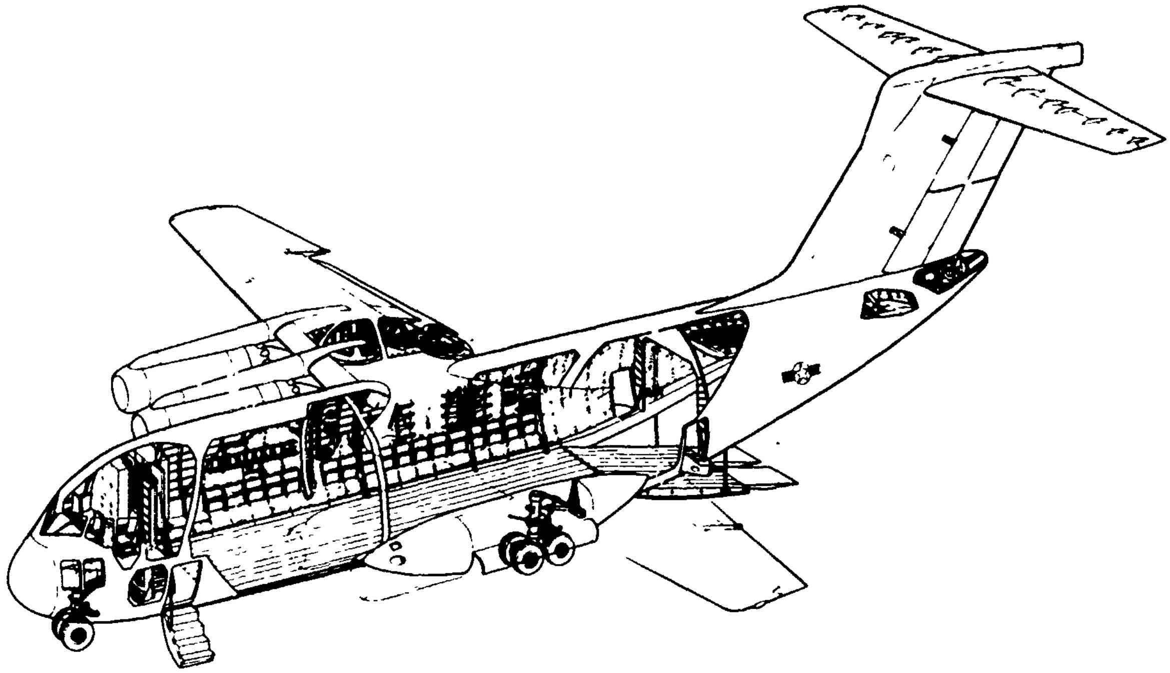


FIGURE 2. CUTAWAY – BASELINE

TABLE 1
BASELINE METAL AND INITIAL COMPOSITE AIRPLANE COMPARISON

	BASELINE	COMPOSITE
WING AREA (SQ FT)	1,740	1,596
THRUST/ENGINE (SLS, POUNDS) (JT8D-17 TYPE)	16,000	13,660
MEW (POUNDS)	98,724	86,880
OEW (POUNDS)	103,234	91,390
TOGW (MIDPOINT) (POUNDS)	150,000	137,400
W/S (MIDPOINT)	86	86
T/W (MIDPOINT)	0.40	0.40
PAYLOAD (POUNDS)	27,000	27,000
DESIGN RADIUS (NAUTICAL MILES)	400	400
MACH NO.	0.70	0.70

CHARACTERISTICS DATA

ITEM	WING (BASIC)	HORIZONTAL TAIL	VERTICAL TAIL
AREA SQ FT	1596	581	412
ASPECT RATIO	7.0	5.0	0.894
TAPER RATIO	0.3	0.45	1.0
SWEEP C/4	$5^{\circ}53'6''$	$4^{\circ}46'12''$	41°
ANHEDRAL	0°	3°	~
THICKNESS % CHORD	13.908% AVE	11%	12%
VOL RATIO	~	1.323	0 1235

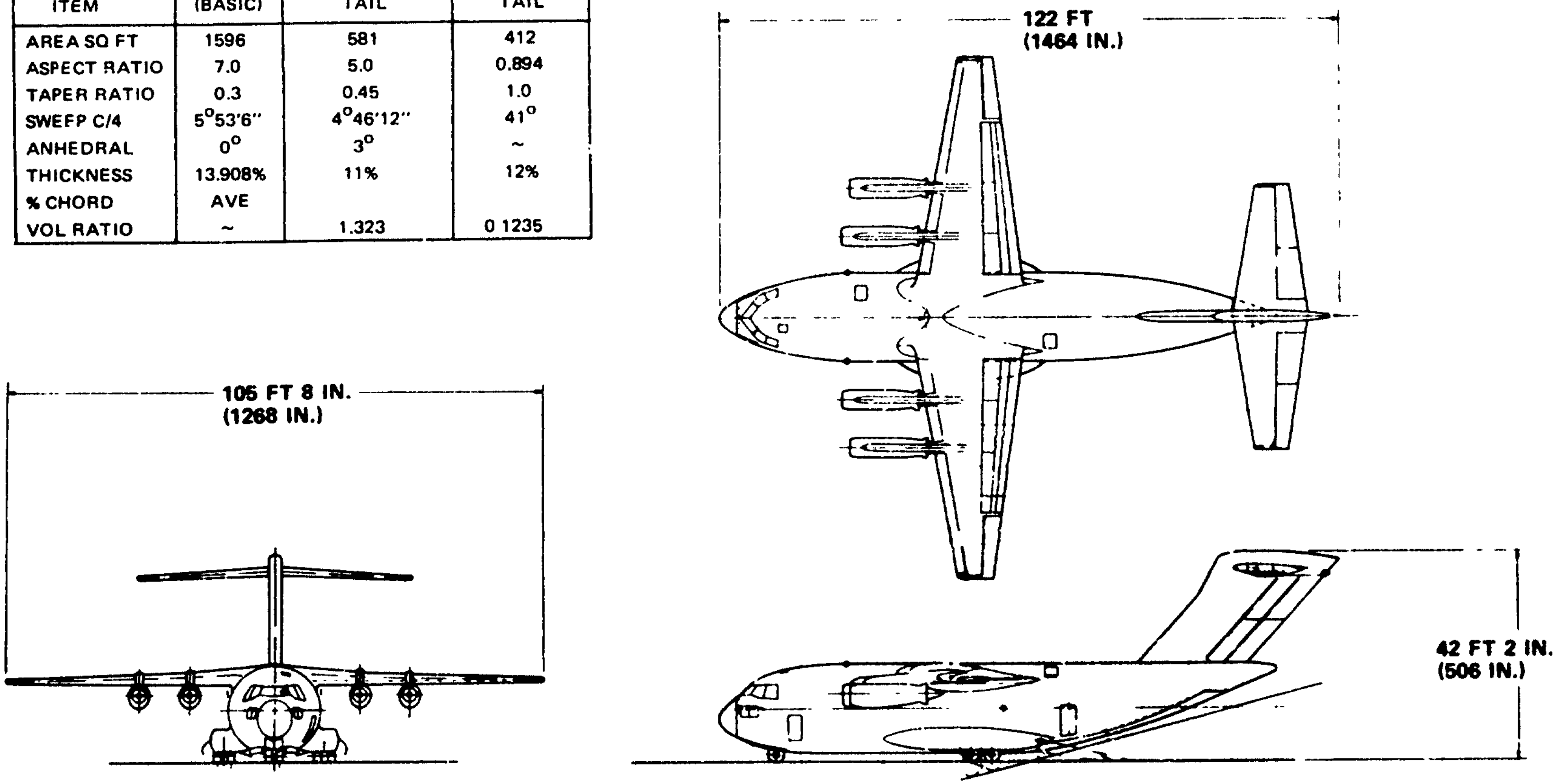


FIGURE 3. INITIAL COMPOSITE AIRPLANE

ramp areas. All-composite construction could not be considered for those high impact and wear areas, but a modest weight saving through selective reinforcement was considered cost-effective.

The lower-cost E-glass-epoxy family of unidirectional tape materials was used for stress concentration relief strips in graphite-epoxy box structures, for rip-stop application in fuselage skins, and it was considered as a filler in the central regions of stability critical solid laminates to significantly reduce average material cost. The difficulty with the latter application was in the differing thermal expansion characteristics of glass and graphite, as for instance in a fuselage cylinder where differential shrinkage forces could cause significant interlaminar preload in the hybrid glass/graphite material. The use of hybrid Kevlar-49/graphite laminates was considered to ease such glass/graphite thermal coefficient problems; however, the availability of "continuous melt" or carbonaceous pitch-based graphite fiber would preclude the necessity of low-cost hybrid materials since the projected $5 per pound cost of pitch-based fiber is lower than projected cost of Kevlar-49 ($10 to $15 per pound). In areas where glass/graphite laminates are feasible, a combination of glass/pitch-based graphite would promise some further economies. In this program, two composite cost categories were established (as explained in Section 6). One represents the average expected Thornel 300 costs and the other represents the lower-cost hybrids and/or the pitch-based graphite product to be available in the near future.

Kevlar-49 was not specified within the program primarily because of its adverse failure characteristics in compression (Reference 3), and secondly because of the potential availability of continuous melt graphite.

E-glass fabric laminates were used in compound curvature areas such as fillets, pods, radomes, electrical isolation areas, and noncritical leading edges of control surfaces such as rudders, elevators, and ailerons.

Composites were not selected for use in true leading edge impact and anti-icing areas such as slats, leading edge flaps, and fixed leading edges because of the added difficulty of erosion protection and thermal conductivity for such areas. Electrically continuous paths for lightning strike charge conduction are also necessary so metal leading edges were used. It was not within the program scope to do detailed cost-effectivity studies of such secondary structure areas although it is recognized composite leading edge designs are possible.

The form of composite material to aid rapid layup was considered dependent upon component size and manufacturing process. Four forms and two thicknesses per ply were identified as useful. See Table 2.

The choice of honeycomb core material for sandwich construction was a subject of study. Electrical discharge machining (EDM) of aluminum core has a low-cost potential for shaping sculptured or tapered sections but large panels will be used in any AMST composite sandwich design. The practicality of EDM shaping large panels was studied and determined feasible. Aluminum core is reported to have a corrosion potential with graphite facings, and high maintenance cost service experience with metal-faced metal honeycomb constructions makes unwise the choice of another potential high maintenance

TABLE 2
COMPOSITE MATERIAL FORMS AND PROJECTED 1977 COSTS

FORM	USE	COST* CATEGORY I ($/LB)	CATEGORY II ($/LB)
A. 12-INCH-WIDE UNIDRECTIONAL PREPREG TAPE	AUTOMATIC TAPE LAYING MACHINERY	35	5
B. STYLE 1050 WOVEN/ UNIDIRECTIONAL PREPREG BROADGOODS	WIDE MATERIAL FOR HAND LAYUP AND TAILORING	25	10
C. SLIT STYLE 1050 PREPREG BROADGOODS	AUTOMATED TAPE WRAPPING AND PULTRUSIONS	31	11
D. CROSS-PLIED (MULTILAYER) PREPREG TAPE	PULTRUSIONS AND RAPID HAND LAYUP	30	15

*NOTE: COSTS PER MATERIAL FORM ASSUMED INDEPENDENT OF MATERIAL THICKNESS. TWO THICKNESSES DEFINED, 0.0055 AND 0.010 INCH PER PLY.

CATEGORY I – THORNEL 300/5208
CATEGORY II – PITCH-BASED : IBER/EPOXY

cost combination such as graphite-faced aluminum core. Lack of data, however, made it difficult to make a case against aluminum core in the face of its obvious cost advantage. A rough cost comparison between EDM-machined aluminum and conventionally-machined nonmetallic Nomex honeycomb core showed the Nomex to be approximately twice the fabricated cost (prior to installation in a part) of the aluminum core. The cost included raw material cost.

As a design philosophy, the use of sandwich construction utilizing the selected corrosion resistant aluminum cores was minimized in favor of solid construction wherever the latter appeared warranted by design considerations.

2.4 DESIGN CONCEPT SELECTION

2.4.1 Rationale for Cost-Design Selection

A choice was made against comparing competing composite designs to metal counterparts on any absolute basis early in the program. The selected basis was to compare significant costs of competing composite designs. Thus, only those major factors are included which contribute directly to cost of one design versus another; i.e., labor, tooling, and materials. Absolute costs of the compared designs were not obtained and at that stage there could be no comparison of composite costs relative to replaced metal components.

Parametric costing procedures involve necessary knowledge of cost estimating relationships, which were not reliably available for composite designs; therefore, the alternate approach using direct estimates was utilized. This is analogous to the production or short-term cost basis making use of, as it does, the major elements of cost to fabricate the component rather than cost to use and service it. The proposed long-term savings in maintenance and repair costs through the use of solid-laminate construction rather than honeycomb sandwich construction did not appear quantifiable as a design selection criterion, so the short-term cost evaluation basis was adopted.

Design-to-Cost - The C-15 production airplane will be a "design-to-cost" airplane; i.e., designed to a specific total production cost. To use target cost as an additional design criterion, the designer must be acquainted with cost elements of his design. Heretofore, cost concepts in design involved general principles for producing lowest cost designs rather than designing to absolute cost. Since cost data on composite designs also are presently limited, techniques to design composite structures to cost were not available for this program.

Cost Effectivity - Cost-effective formulations assume use of equivalent data bases from which costs of conventional metal and new design components are defined. The data, however derived, are used in the form of dollars per pound. Cost per pound is known for conventional structure for which a historical data base exists. There was no comparable data base for composite structures sufficient to allow reliable comparisons of composite designs.

These design-to-cost and cost-effectivity considerations suggested that the most reliable design selection technique would be the use of comparative labor, tooling, and material cost estimates for the several designs. This approach simulated the industrial engineering standards estimating technique.

2.4.1.1 Composite Component and Subcomponent Definition

For purposes of the study, airframe components were classified into the following three groups with varying emphasis placed on each group:

Group I - Major Primary Structure

The greatest payoff was expected when aircraft resizing was considered, so that principal attention is paid to components which develop the greatest weight saving when designed for use of composites. This criterion narrowed consideration to the following components:

- Wing Structural Box
    - Covers
    - Substructure
- Fuselage
    - Basic shell structure
    - Complete structural shell
- Empennage structural boxes
    - Covers
    - Substructure.

These components comprise 50 percent of the airframe weight for the study baseline aircraft. Special emphasis was given the selection of lowest weight and lowest cost components for Group I primary structure.

Group II - Major Secondary Structure

Components in this group have less potential for weight saving, have special design problems, or were outside program scope due to budgetary and time limitations. They were therefore studied on a lesser level of detail than Group I components. Group II consists of the following components:

- Wing flaps
- Floors and floor supports
- Landing gear pods
- Fairings

- Doors and cargo loading ramp
- Engine pylons.

Group III - Secondary Structure

Group III components include:

- Leading edge control surfaces
- Fixed leading edges
- Ailerons and spoilers
- Rudders and elevators
- Engine nacelles.

The program charter included no special emphasis on secondary structure; however, some of these components can yield significant weight saving when properly designed in composites. Also, fabrication hours per pound (i.e., dollars per pound) were found to be significantly higher in the baseline metal secondary structure, offering great opportunity for overall cost savings in those areas. Accordingly, the several areas of Group III structure were considered for composite design within program limitations.

2.4.2 Wing Box Concept

Following a preliminary design concept evaluation (Appendix B), three wing box concepts were subjected to a preliminary but detailed relative labor and materials estimation to determine the design with the lowest cost potential. The candidates were two multirib concepts and a truss-web concept. The two multirib candidate concepts were a honeycomb panel design and a solid-laminate-stiffened design (Figures 4 and 5). The truss-web concept (Figure 6) was an improved version of several considered during the initial evaluation described in Appendix B.

Figures 4, 5, and 6 show these designs in the form of cost estimating units of arbitrary size. The size and detail were selected to represent the major fabrication and assembly differences. The detail thicknesses specified were compatible with load requirements at midspan on the wing.

The two multirib designs (Figures 4 and 5) were selected to define cost-weight differences between solid stiffened and honeycomb panel design. Ribs are bonded with cocured angles to lower cover and spar webs in the sandwich design and the upper cover is attached with bolting and bonding, the assembly procedure being similar to the graphite A-4 stabilizer (Reference 4). The solid laminate multirib design has rib attach channels bonded to both covers and has angles or T's bonded to the spars, to which the integrally stiffened ribs are room temperature bonded and mechanically fastened.

The composite covers, end webs, and integral stringers of the truss web (Figure 6) are of solid-laminate-stiffened construction, consistent with

ASSEMBLE ALL BUT UPPER COVER BY INTERNAL VACUUM BAGS AND COCURE AND BOND ANGLES

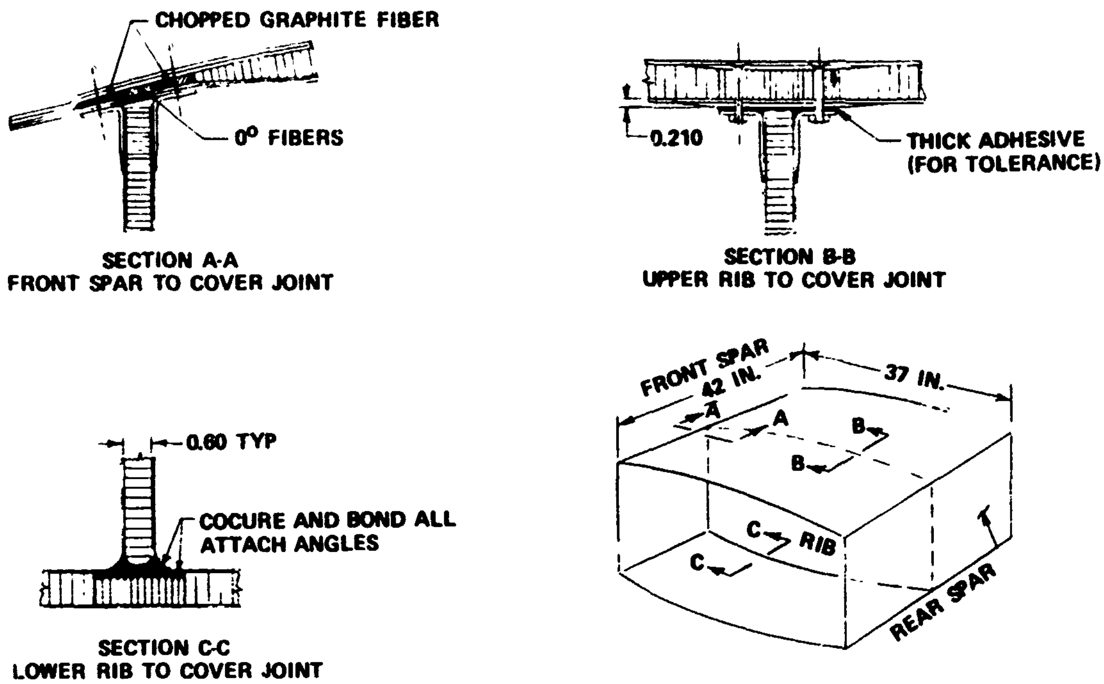


FIGURE 4. HONEYCOMB MULTIRIB COST COMPARISON BOX

ASSEMBLED WITH ROOM TEMPERATURE ADHESIVE (POSTCURED) AND FASTENERS.
"HAT" SECTIONS USED BECAUSE OF AVAILABLE DESIGN AND WEIGHTS.

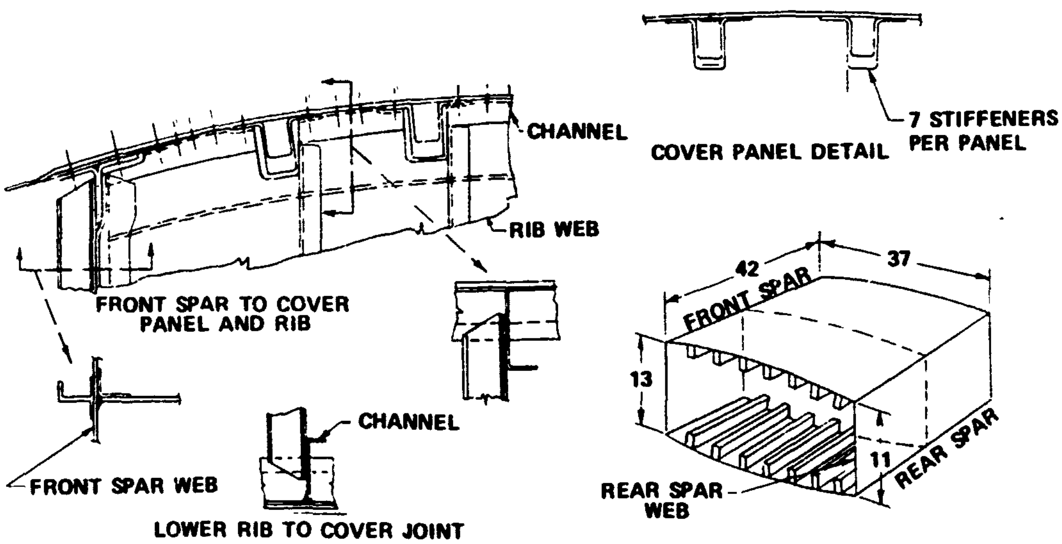
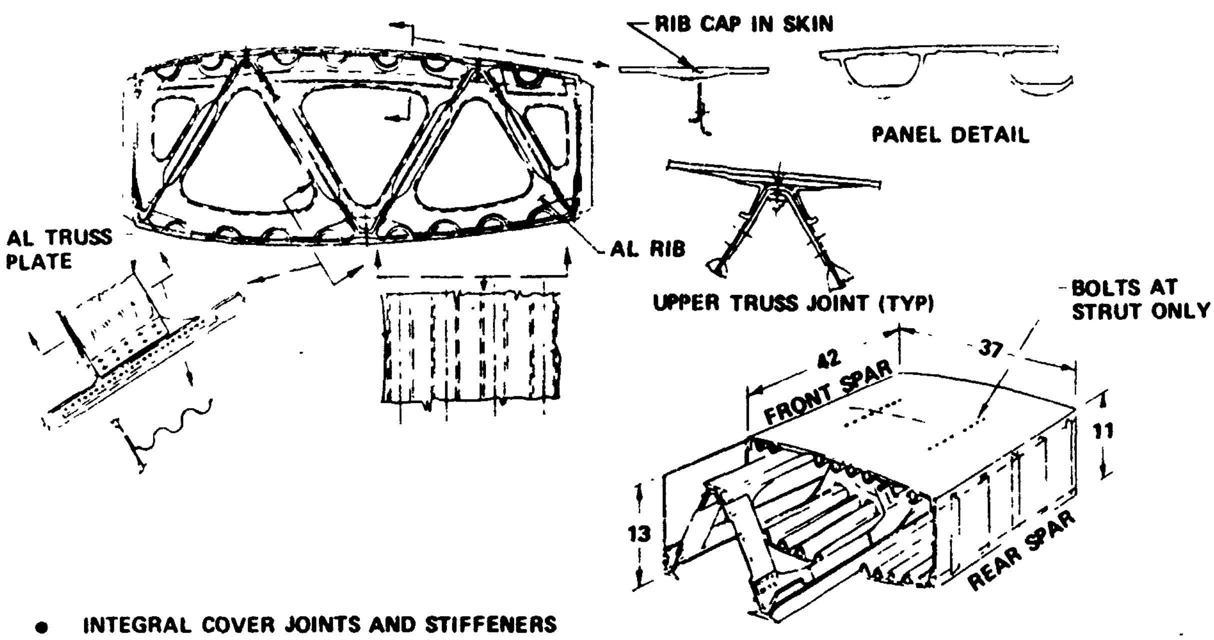


FIGURE 5. SOLID LAMINATE MULTIRIB COST COMPARISON BOX



- INTEGRAL COVER JOINTS AND STIFFENERS
- ALUMINUM SUBSTRUCTURE

FIGURE 6. TRUSS WEB COST COMPARISON BOX

judgment that the contoured and tapered honeycomb stiffening previously seen in these designs is too costly and the presence of honeycomb webs in the fuel tank is environmentally suspect. Integral "derby hat" stiffening is shown as on the covers since it was believed to be the most efficient and lowest cost cover stiffening concept from the initial evaluation.

Aluminum substructure was incorporated in the truss web box on the anticipation that formed metal sheet in the generally more expensive substructure area would be less costly to fabricate than variable corrugated and cutout solid-composite truss webs or variable thickness honeycomb plate webs (Reference 5). Program timing did not allow a separate trade study on the best way to fabricate low-cost truss web plates. Hydroformed aluminum panels were also specified to provide chordwise support stations for the solid stiffened covers. Although the aluminum substructure elements might cause a small increase in long-term operating cost due to substructure weight increases over all-composite truss webs, it was felt a short-term cost advantage would enable truss web to compete effectively with multirib design.

With the assistance of Manufacturing Research and Development, actual fabrication hours for T-100 (100th unit) were obtained as well as estimates of initial tooling cost. Materials cost was based on weight estimates derived from the design dimensions.

Configuration identification letters A, B, and D were assigned the three box estimating units (Figures 4, 5, and 6, respectively) as shown in Table 3. Table 3 also shows two additional truss web configurations, C and E, added after examination of the initial cost and weight results from A, B, and D. Of the truss webs, it appeared that the C truss web would combine lower weight and lower cost substructure and covers than D, while E configuration (with chordwise stiffening) was introduced to eliminate the necessity of rib-like supports for the covers and to provide an easier cover design for chordwise load introduction. The sandwich panels of C configuration are self-supporting and also require no rib supports. Cost and weight estimates were rationally extrapolated to C and E configurations.

Table 4 presents the relative weights of the five final selections based on the dimensions of the three cost estimating boxes. It will be noted that the C truss web has the lowest weight covers; however, the metal truss webs, the heavier "stringers" for cover support, and the unsupported honeycomb shear panels of C configuration do not allow as much weight saving in substructure with respect to the A multirib as the optimum truss web of Reference 5. The heavier covers and substructure of the B multirib with respect to unit A are a consequence of solid stiffening, all elements of which in this moderately loaded structure are not operating at maximum stress level. Definition of weights for the solid chordwise stiffened covers of the E truss web proved difficult from a structural analysis point of view. It was estimated the covers might be 1.8 times as heavy as the spanwise stiffened multirib B covers and this was expected to produce unacceptable recurring costs for E configuration composite materials.

Relative costs are shown in Table 5. Cumulative costs for 100 units were obtained on a basis of recurring materials and labor, and nonrecurring tooling estimates.

The solid-laminate multirib B with conventionally shaped hat stiffeners shows 7-percent less cost than the sandwich multirib A when maximum automation was considered (pultrusion) to form the stiffening elements. With multiple-stiffener, hand-layup techniques the labor was not sufficiently reduced to overcome the much increased composite materials cost of the solid stiffened construction. Composite costing $20 per pound was used in these comparisons. Tooling differences, being nonrecurring, were not a big factor.

The C truss web with sandwich shear webs and covers shows 16-percent less cost (Table 5) than the D truss web with solid laminate covers and shear webs due primarily to the higher composite material usage in unit D.

Selection of the C truss web over the multirib designs appears warranted, though it is of equal weight, by its apparent significantly lower cost than either the A or B multiribs. This was because of both material and labor cost reductions through the introduction of low-cost aluminum substructure and cover weight reductions with respect to multirib. Aluminum ribs for the multirib cases were not considered because of the problem of thermal mismatch between graphite covers and aluminum ribs over 100- to 120-inch chord distances. Thermal mismatch is not a problem for the aluminum truss configurations because of geometry.

TABLE 3
FINAL WING AND EMPENNAGE BOX CONCEPTS

CONFIGURATION	DESIGN PROBLEMS
A. HONEYCOMB SANDWICH PANEL MULTIRIB	HONEYCOMB IN FUEL AREA FUEL VOLUME REDUCED VULNERABILITY, RELIABILITY, REPAIRABILITY
B. SOLID LAMINATE MULTIRIB	UNKNOWN REPAIRABILITY
C. TRUSS WEB – SANDWICH COVERS AND ALUMINUM STRUTS	HONEYCOMB FUEL PROXIMITY FUEL VOLUME REDUCED VULNERABILITY, RELIABILITY, REPAIRABILITY
D. TRUSS WEB – SOLID LAMINATE COVERS WITH SPANWISE STIFFENING, ALUMINUM STRUTS AND RIBS	CHORDWISE LOADS REDUNDANT COVER SUPPORTS
E. TRUSS WEB – CHORDWISE STIFFENED SOLID LAMINATE COVERS WITH ALUMINUM STRUTS SUBSTRUCTURE	INHERENTLY HEAVY COVERS (DIFFICULT ANALYSIS) UNKNOWN REPAIRABILITY

TABLE 4
RELATIVE WEIGHTS – WING BOX CONCEPT FINAL SELECTIONS

	MULTI-RIB		TRUSS WEB		
	A	B	C	D	E
			NO RIBS		NO RIBS
COVERS	1.00	1.13	0.96	1.20	1.89
SUBSTRUCTURE	1.00	1.13	1.09	1.14	1.12
COVERS PLUS SUBSTRUCTURE	1.00	1.13	1.00	1.18	1.63

NOTE: ACTUAL WEIGHTS NORMALIZED PER DATA ROW

TABLE 5
RELATIVE COSTS OF WING BOX ESTIMATING UNITS

	MULTIRIB		TRUSS WEB		
CONFIGURATION	A	B	C	D	E
FABRICATION LABOR (OPTIMUM FABRICATION)	1.46	1.13	1.00	1 02	1.41
MATERIAL	1.22	1.63	1.00	1.41	2.07
TOOLING (INCLUDING TOOLING MATERIAL)	1.00	1.41	1.95	2.90	2.50
TOTAL (NORMALIZED)	1.35	1.28	1.00	1.16	1.61

NOTE: BASED ON 100 UNITS

The subsequent introduction, during detail design (Paragraph +. 1), of the solid-laminate stiffened shear webs to replace the honeycomb shear webs was in the interest of further labor cost reduction with respect to sandwich panels. The estimated added weight and materials cost, in such solid webs, was deemed not sufficient to change the configuration selection, although a new estimate was not obtained at that time.

2.4.3 Fuselage Concepts

Figures 7, 8 and 9 present the fuselage concept candidates in the form of cost estimating units. As with the wing boxes, fabrication man-hour estimates were obtained for the three fuselage shell sections. Corresponding weight estimates for the sections, including one circumferential end joint, were obtained as shown in Figures 7, 8, and 9. Relative weight and relative labor and material costs for 100 units are shown in Table 6. Tooling cost estimates were not obtained so the comparison was made on a basis of factored labor and material costs ($20 per pound composite) only. The costs of the structural foam core arch frame design were not considered because of the obvious weight penalty. The foam density had been selected to provide a high heat distortion point to allow cocure and bonding of the composite.

The indicated costs of the isogrid and honeycomb shell were so similar that other factors, such as weight and environmental resistance (which are apparent long-term cost factors), were considered in the selection. The isogrid fuselage concept was the recommended selection because of weight and cost reduction potential.

As indicated in Table 6, the all-solid isogrid offers a chance to dramatically reduce the cost of materials by interspersing inexpensive glass roving in the central portion of the grid rib height without a significant change in labor cost. The concept also offers lower cost and weight penalties than sandwich to accommodate:

- Systems attachment
- Reinforcement for local loads or discontinuities from cutouts
- Provision for failsafe.

Circumferential bolted joints at cylinder edges are more forgiving of assembly tolerance problems with a mating cylinder than is the double plate butt splice of the sandwich cylinder. Isogrid allows use of an efficient V-joint analagous to the truss web wing concept. Isogrid has fewer failure modes and therefore offers a reduced cost-test program for structural verification.

2.4.4 Empennage Box Concepts

The same truss-web concept was selected for the horizontal stabilizer box as for the wing box on the basis of comparisons of weight and cost made in Tables 4 and 5. The truss web adapts well to the eccentric hinging requirements for the elevator. Also, four machined, segmented bulkheads in the 2-foot-wide center box area of the stabilizer may be replaced with a single centerline bulkhead and wide truss webs which not only support the covers but also redistribute stabilizer pivot reaction loads and actuator loads.

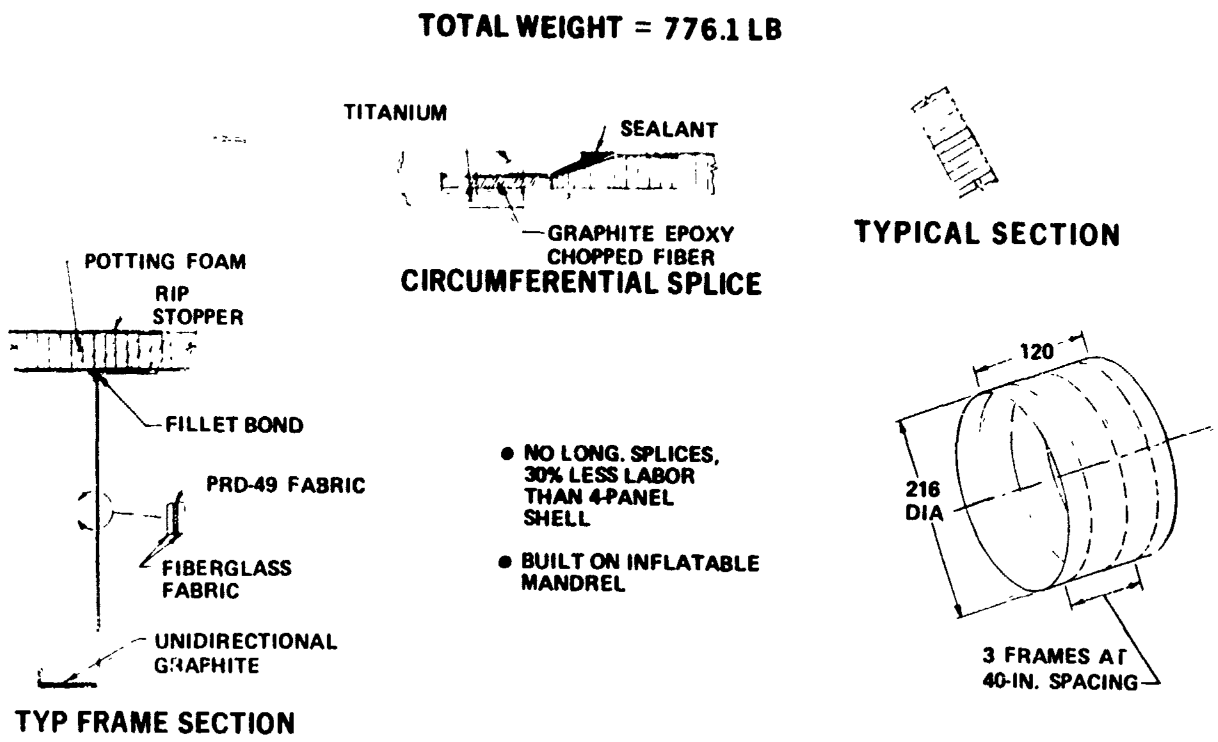


FIGURE 7. FUSELAGE COST COMPARISON SECTION – THICK HONEYCOMB

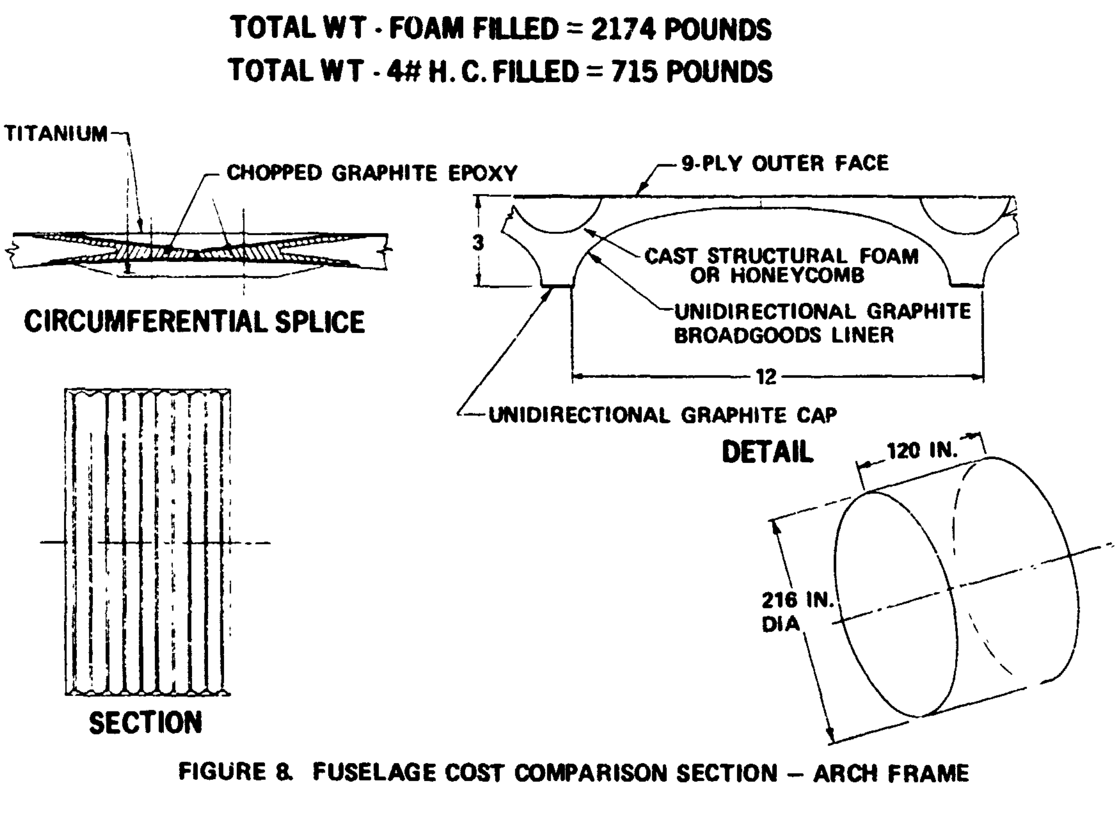


FIGURE 8. FUSELAGE COST COMPARISON SECTION – ARCH FRAME

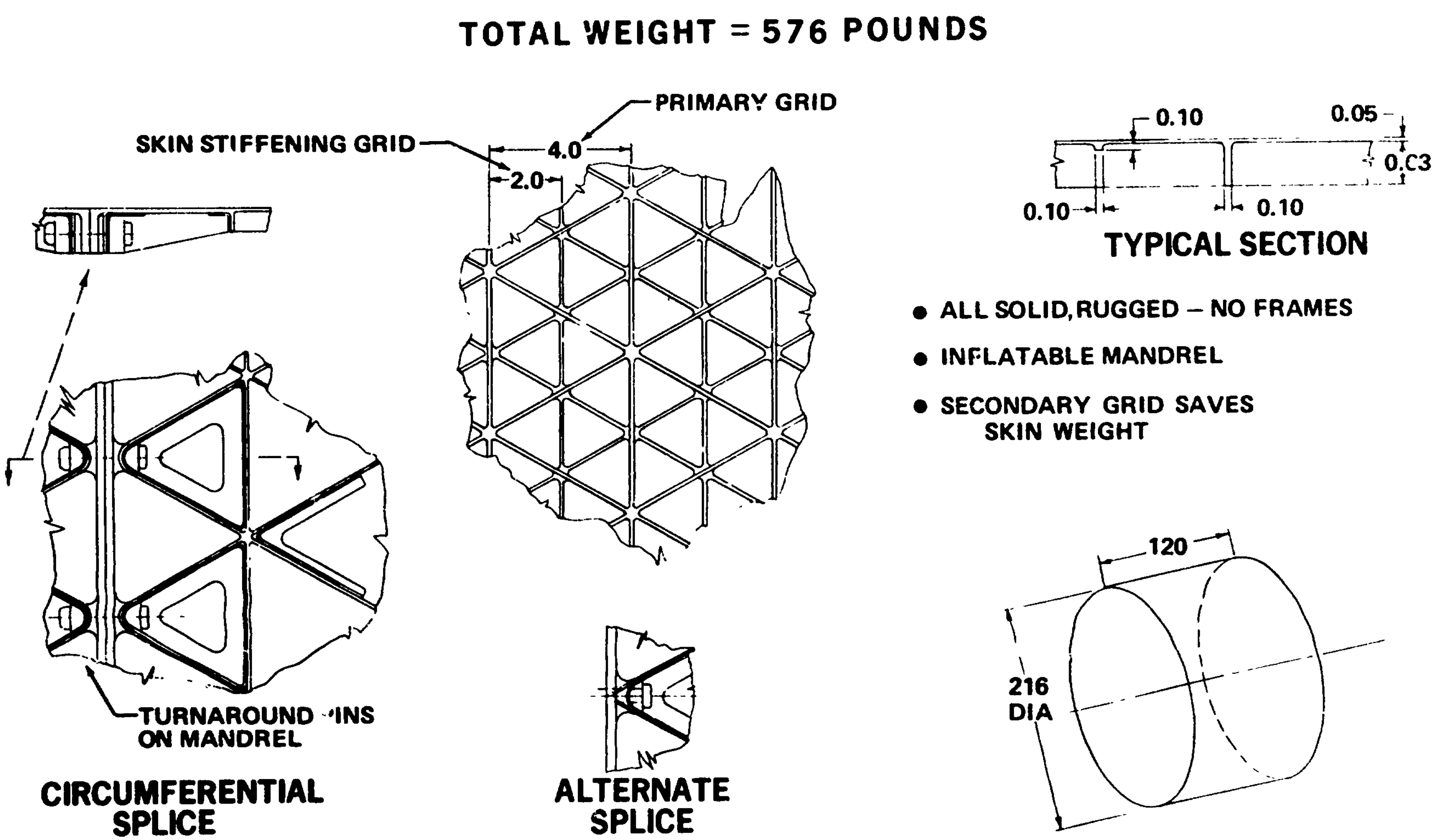


FIGURE 9. FUSELAGE COST COMPARISON SECTION – ISOGRID

TABLE 6
RELATIVE WEIGHT AND COST EVALUATION – FUSELAGE CONCEPTS

CONCEPTS	RELATIVE WEIGHT	COSTS (100 UNITS)		
		MATERIALS[1]	LABOR	TOTAL
HONEYCOMB SANDWICH SHELL WITH FRAMES	1.35	1.16	1.00	1.06
ARCH FRAME, 4_{pcf} HONEYCOMB	1.24	1.19	1.26	1.17
ISOGRID (ALL GRAPHITE)	1.00	1.00	1.10	1.00
ISOGRID (GLASS/GRAPHITE)[2]	1.07	0.35	1.10	0.62

NOTES: 1. GRAPHITE AT $20/POUND HONEYCOMB, GLASS, ADHESIVE, METAL, ETC.
2. TREND ONLY – THERMAL DESIGN NOT FEASIBLE.

The large variety of configuration alternatives did not appear to be available for the vertical tail de oncept selection. The metal baseline configuration is multirib with tv.. . spars and added spars (five total) at the fuselage/fin intersection .o aid load transfer through the pressurized fuselage shell in the form of concentrated loads. These spar cap loads react in shear on fuselage slant bulkheads. This basic vertical tail design was retained in the composite airplane although sandwich panels were utilized rather than solid stiffened panels to lower composite material utilization.

2.4.5 Secondary Structural Concepts

As explained in Paragraph 2.3, fixed and movable leading edges of the baseline metal design were retained. Engine nacelles require special study in the areas of acoustic fatigue and temperature resistance and are, therefore, more appropriately the subject of a separate program. Accordingly, no effort was expended on composite design and baseline metal nacelle weights and costs were retained. Engine pylon stiffnesses are critical for wing flutter stiffness and fire resistance is a crucial design concern; therefore, it was judged that proper consideration could not be given to pylon design and again the baseline metal design was retained. The same comments apply to the wing flaps which are subject to approximately 550°F maximum service temperature.

The remaining trailing-edge control surfaces, ailerons, spoilers, rudders, and elevators, were deemed a high payoff design area in terms of cost and weight. The availability of a composite control surface design concept from another program (Reference 6) enabled inclusion of these components in weights and cost evaluations. In the extensive trade studies leading up to design selection for the referenced rudder program, many alternate designs were considered as shown in Figure 10. An emerging fabrication concept that allowed integrally molded covers and substructure promised maximum structural reliability for the simple multirib concept. Along with reliability and lowest weight, low cost guided the design selection, Figure 11. All parts shown on Figure 11, except leading edge, drive fittings, tip, and trailing edge angles, are integrally molded together, greatly reducing the effective number of parts and eliminating the higher cost associated with sandwich construction. The selected composite multirib control surface design is essentially the same as for the metal baseline except ribs are without lightening holes. Since rib and spar flanges taper integrally into the skins, the fingered, bonded acoustic fatigue doublers, which are typical of the metal design, may be eliminated. Fastener holes, except for hinge fitting attachment, are also eliminated. The feasibility of using the unstiffened skin design approach has been successfully demonstrated with subcomponent tests on the DC-10 Rudder Program (Reference 6). See Figure 12.

The remaining Group II and III secondary structure, floors and floor supports, doors, and cargo loading ramp, etc., were considered in the detail design phase, Paragraph 4.1.

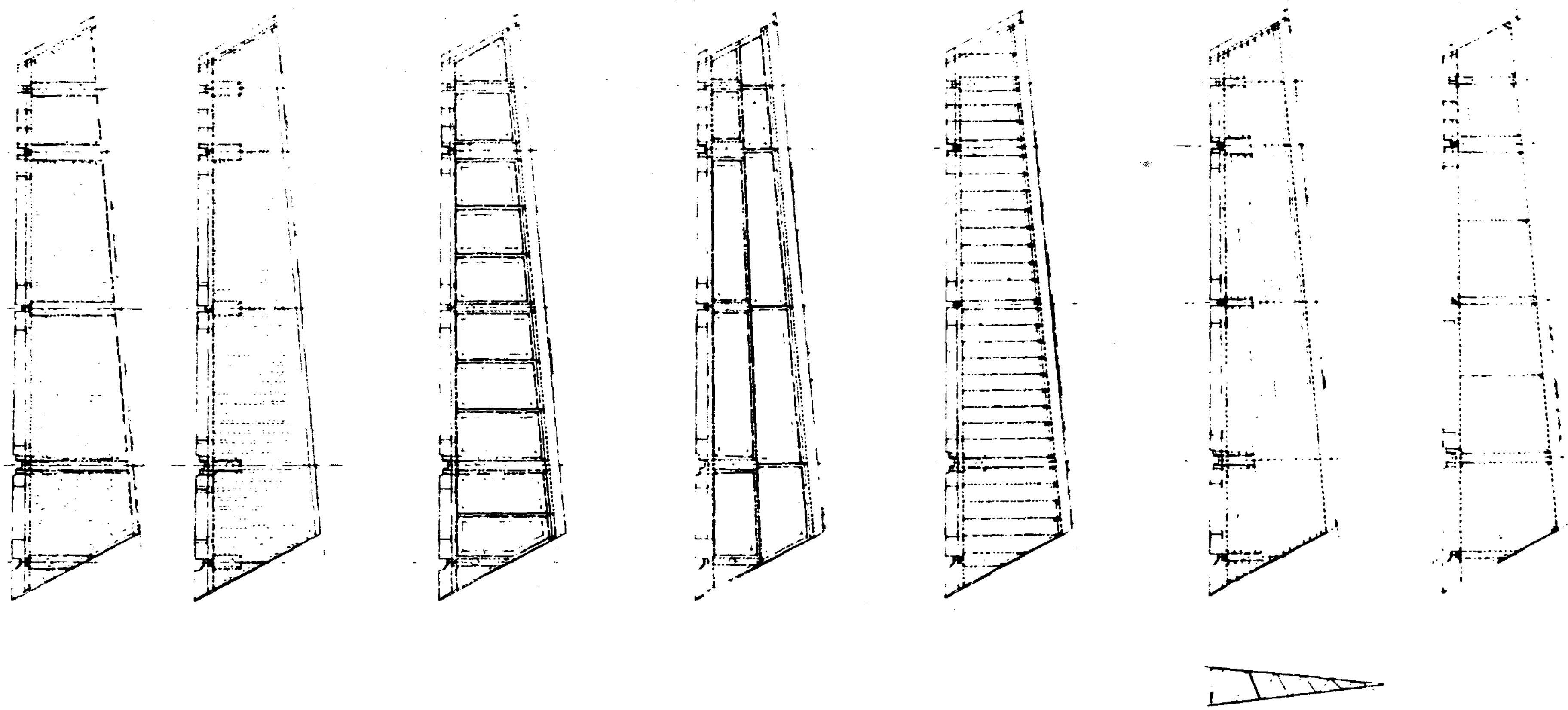


FIGURE 10. CANDIDATE STRUCTURAL CONFIGURATIONS – CONTROL SURFACE

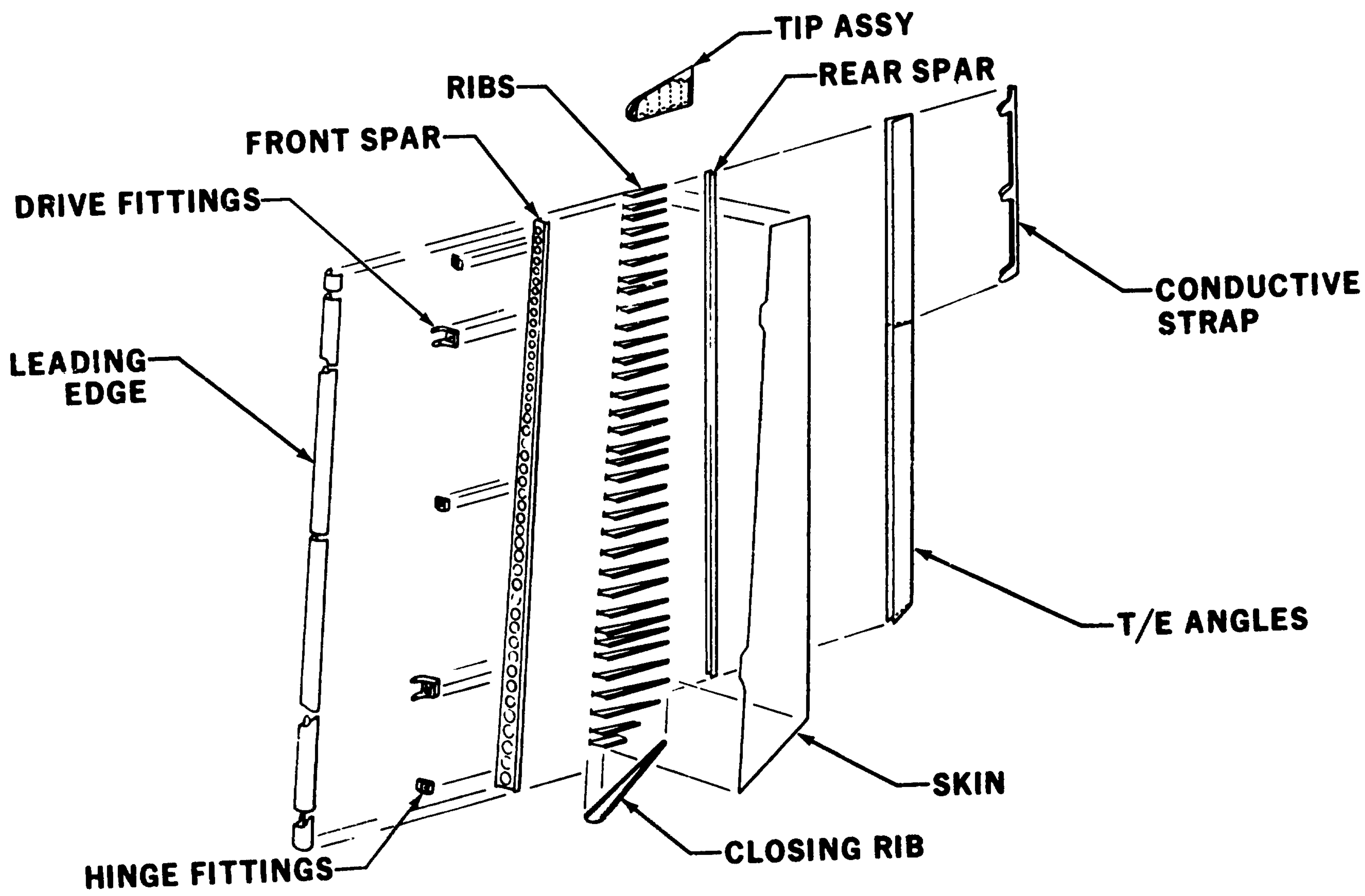


FIGURE 11. STRUCTURAL ARRANGEMENT FOR GRAPHITE RUDDER

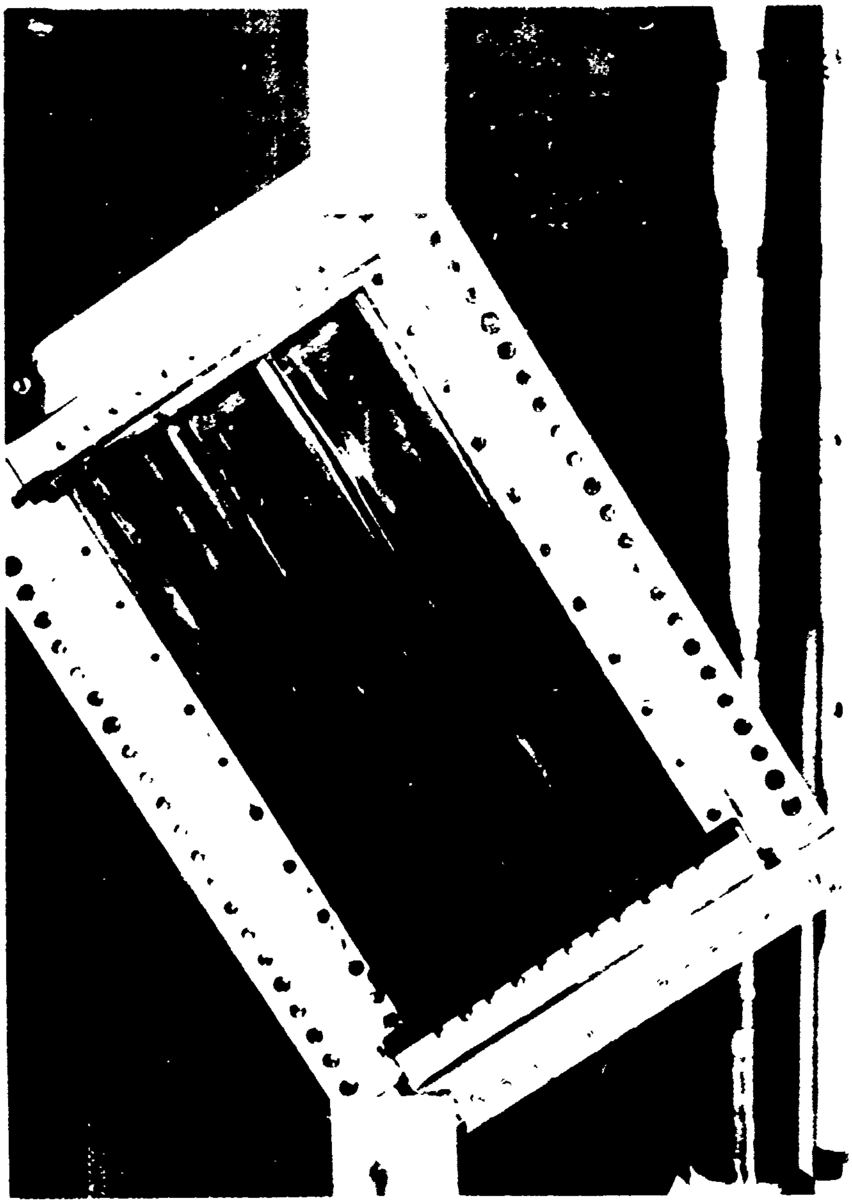


FIGURE 12. SHEAR PANEL UNDER LOAD

2.4.6 Stiffened Panel Evaluation

To aid detail design selection, a cost/weight trade study was conducted for various types of solid graphite-epoxy, integrally-stiffened and honeycomb panel designs. Appendix B describes the initial survey of panel concepts. From the 17 wing cover and 12 substructure panel concepts evaluated in the preliminary concept selection study, three finalists emerged. They were:

- J-stiffened solid laminate panel
- "Derby hat"-stiffened solid laminate panel
- Constant thickness sandwich panel with aluminum honeycomb core

Table 7 shows relative weights and preliminary fabrication costs consisting of recurring labor and material. Detailed tooling labor and tooling materials were not estimated but a relative ranking was estimated to support the other cost elements. Figure 13 shows dimensional data for two of the three panel types. Design loads were 12,000 pounds per inch compression and 4000 pounds per inch shear, representative of the upper cover at Wing Station 214. The panels were not designed for pressure.

The derby hat was shown to be an efficient solid-laminate stiffening concept. Its labor cost can be less than the sandwich panel although the added cost of the solid graphite with respect to the sandwich panel is a disadvantage. The derby hat could be further optimized in specific applications by taking advantage of the contribution by the continuous inner sheet to panel shear stiffness and shear strength. Although the derby panel was not selected for cover panel stiffening, it was incorporated in spanwise shear webs of the wing and empennage.

The apparent keys to successful low-cost usage of solid laminate stiffening are material cost (lower than $20 per pound used in this estimate) and labor/material cost ratio. Prototype panel estimates were included as well as T-[illegible] production estimates to assess any change in ranking due to numbers of [illegible]s. A change in ranking did not occur, since nonrecurring tooling costs were not included, and even though there was a large change in labor/material ratio between T-1 and T-100 estimates. The simple sandwich panel was therefore indicated as a low-cost selection; however, additional factors of complex edge closures, through fasteners, and sandwich tapering were not included. The sandwich panel in this trade study used "as purchased" constant thickness aluminum core.

If the sandwich panel used conventionally machined and expanded tapered core, rather than the core being used in the as-purchased constant thickness condition, the relative T-100 labor cost of the sandwich panel would change to 2.20 and the total cost to 1.43, thus exceeding the derby hat cost. Use of machined tapered Nomex core rather than the constant thickness aluminum core raises the sandwich panel total relative cost to 1.48. This accentuated the potentially cost competitive nature of some solid-stiffened panel designs.

TABLE 7
STIFFENED PANEL RELATIVE WEIGHT AND COST COMPARISON

	RELATIVE WEIGHT	RELATIVE COSTS – PRODUCTION: MATERIAL IN PART	LABOR	TOOLING (RANKED)	TOTAL
• CONTINUOUS INNER SHEET DERBY HAT	1.11	1.98	1.00	(2)	1.30
• J-STIFFENED PANEL	1.13	2.03	1.64	(3)	1.50
• HONEYCOMB SANDWICH	1.00	1.00	1.24	(1)	1.00
		RELATIVE COST – PROTOTYPE PANEL			
• DERBY HAT	SAME AS ABOVE	SAME AS ABOVE	1.18	–	1.25
• J-STIFFENED	SAME AS ABOVE	SAME AS ABOVE	1.49	–	1.55
• HONEYCOMB SANDWICH	SAME AS ABOVE	SAME AS ABOVE	1.00	–	1.00

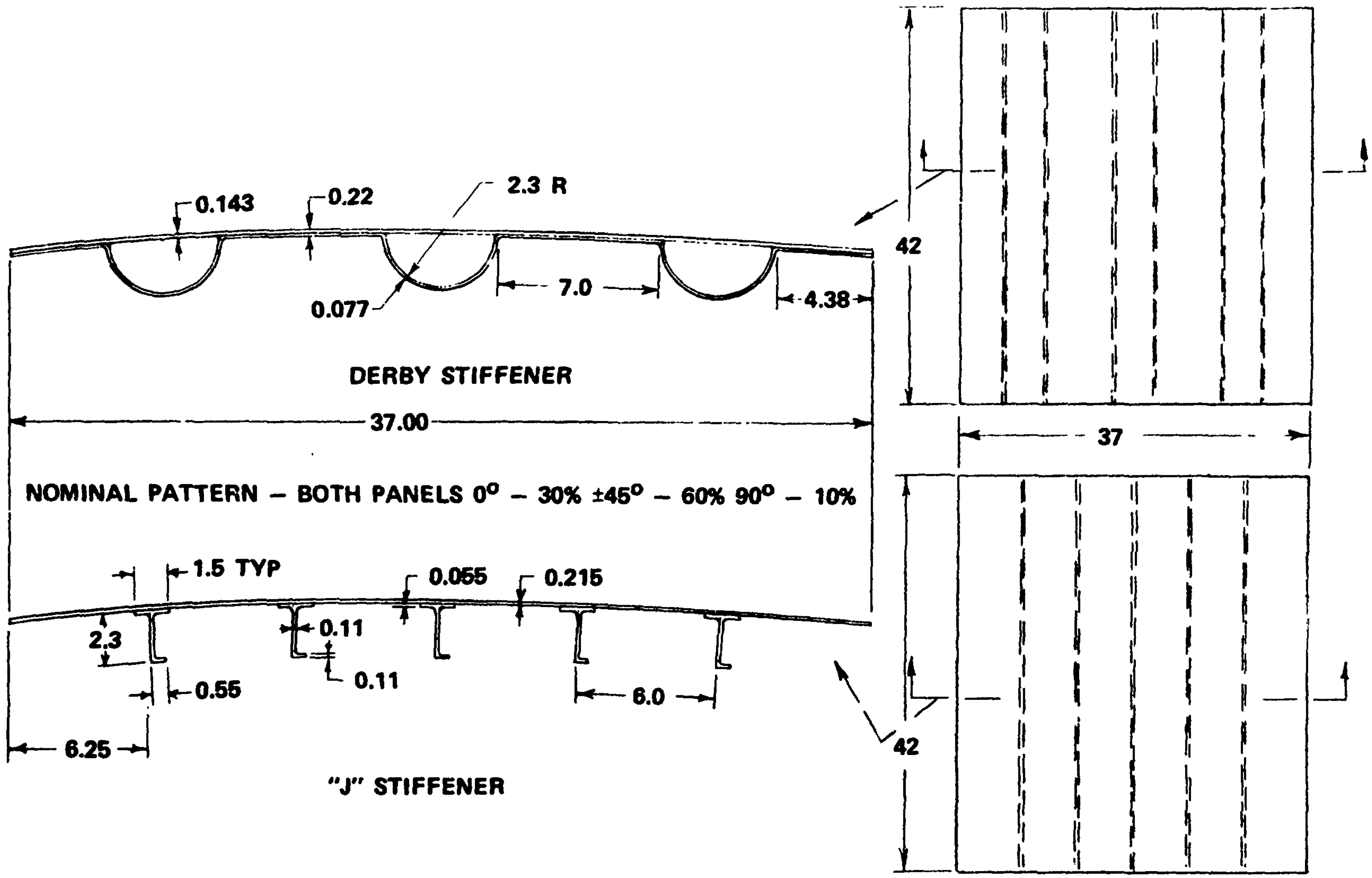


FIGURE 13. COST COMPARISON PANELS – SOLID LAMINATE

SECTION 3
CONFIGURATION PARAMETRIC SENSITIVITY ANALYSIS

3.1 EFFECTS OF PERTURBING AERODYNAMIC PARAMETERS

The primary criterion for selection of the composite AMST wing geometric characteristics, as it was for the baseline metal wing, was to minimize aircraft initial cost. Having selected portions of the airframe in which composite materials are to be utilized, minimum aircraft weight results in minimum aircraft initial cost. Minimum aircraft weight generally corresponds to minimum aircraft cost for given construction and materials. Since there was no specific Mach number requirement, the prototype wing geometry selection was not based on the tradeoff between aircraft performance and weight but rather only on minimum weight. Accordingly, the use of composite materials instead of metal affects only wing size, not the geometry. Selection of wing geometry is discussed below.

3.1.1 Aspect Ratio

Figure 14 shows the effect of aspect ratio on aircraft weight. There is essentially no difference in gross weight for aspect ratios between 7 and 9. Factors such as lateral control response, aeroelastic effects, structural dynamics, and aircraft overall dimensions favor the lower aspect ratios. An aspect ratio of 7 was selected for these reasons.

3.1.2 Wing Sweep

Reducing wing sweep reduces weight as shown in Figure 15. A wing sweep of 5.9 degrees was selected for the prototype in order to provide a straight flap hinge line perpendicular to the airstream. The small reduction in wing weight associated with zero-sweep angle would be more than offset by the higher flap attachment structure weights and increased structural cost.

3.1.3 Wing Thickness Ratio

Increasing wing thickness ratio (t/c) will decrease weight, as shown in Figure 15, and increase fuel volume. An average t/c = 0.139 was selected to provide adequate fuel volume for ferry mission capability. Higher t/c does not result in an appreciable additional weight saving but does degrade maximum speed capability. The 0.139 thickness ratio allows cruise at Mach numbers greater than 0.70.

3.2 FUSELAGE DIAMETER

The selection of an isogrid concept for the fuselage implies that the fuselage diameter may be reduced since frames are theoretically not required for shell general stability. Investigation of the geometric relationships between cargo box size and fuselage diameters, however, revealed that the basic fuselage diameter was not selected to provide necessary frame space around the cargo box envelope, since it is already bigger than it needs to be to fit the 11.3- by 11.7-foot box size. Weight penalties can total 3 to 4 percent of fuselage primary structure for the minimum diameter about a given box size (Reference 7) and since the associated drag increases and decreases result in

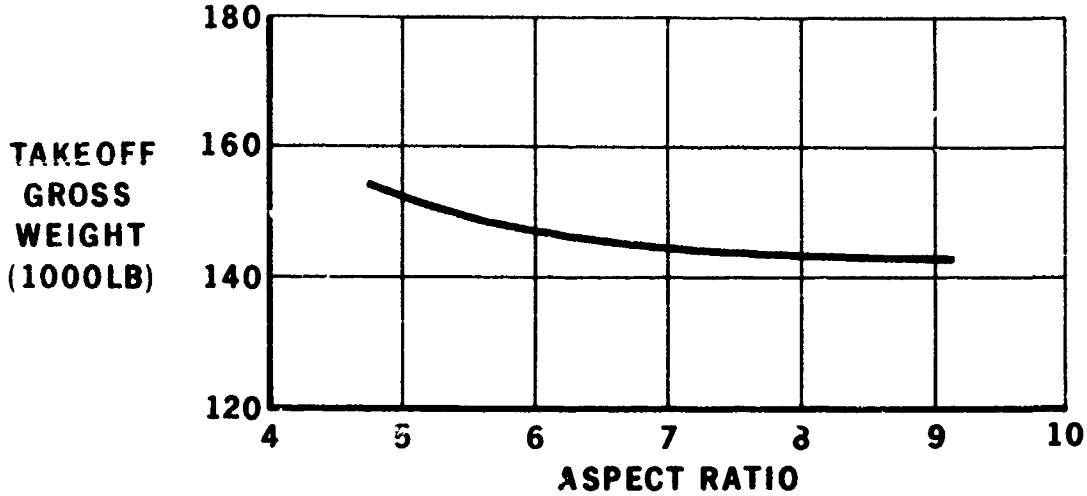


FIGURE 14. EFFECT OF WING ASPECT RATIO ON WEIGHT

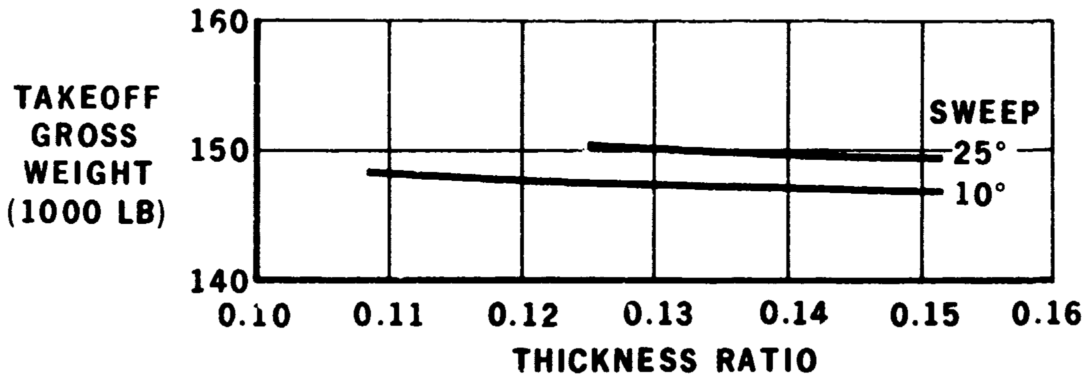


FIGURE 15. EFFECT OF WING SWEEP AND THICKNESS RATIO ON WEIGHT

negligible change in drag, it was concluded that composite airplane resizing would not include fuselage geometry changes.

Smaller diameter fuselages incur weight penalty increments from several sources. A wing/fuselage height effect incurs weight increments for smaller diameters due to larger wing/fuselage fairings, increased bending moments in the major frames for wing and landing gear support, and increased fuselage carry-through structure at fuselage centerline. The structural clearances for vehicles cresting at the top of the loading ramp and also pallet air drops determine the shape of the aft fuselage. The smaller diameter fuselage shapes have to be expanded locally to provide this clearance and they incur weight penalties due to reflexes and flattened areas in the pressurized shape. The landing gear pods also enlarge with respect to a smaller fuselage.

SECTION 4
DETAIL DESIGN

4.1 STRUCTURAL DESCRIPTION

The composite design concepts for the primary structure (wing box, fuselage shell, and empennage boxes) and for selected secondary structure (trailing edge control surfaces) were sized to meet the same criteria as the baseline aircraft. Structures remaining as conventional metal design include landing gear, slats, and movable leading edges (which are subject to hail, erosion, and anti-icing requirements), and certain major frames, forgings, and fittings. As explained in Paragraph 2.4.5, Secondary Structural Concepts, certain othe baseline metallic structures were retained, such as pylons, nacelles, and wing flaps. For reference, Figure 16 is included to show the baseline metal structural arrangement.

With the selection of composite design concepts for various areas of the airframe, manufacturing cost estimating (MCE) drawings were generated. These drawings were developed to provide the essential structural details which have significant impact on manufacturing costs. They include such detail as ply pattern and thicknesses. The necessary level of detail and the composites estimating methodology for a formal "bottoms up" cost estimate were first developed and exercised on a prior composites cost benefits program (Reference 2).

Detailed drawings developed for cost estimating purposes are in Appendix C, Manufacturing Cost Estimating Drawings. These drawings are referred to in the following text.

4.1.1 Wing

Detailed design of the box section only was conducted, and the drawing for the box is given in Figure C-1 of Appendix C. The structural box is a truss web configuration with sandwich covers, solid-laminate front and rear webs, and beaded aluminum truss plate substructure. The covers contain solid-laminate, K-section graphite pultrusions acting as cover supports between truss web stations, and as spar caps and connections to the front and rear shear webs. The truss web plates are bonded and mechanically fastened to aluminum V-plates which in turn nest into the K-sections. These are mechanically tied through solid-laminate portions of the covers to resist internal pressure. The truss plate stations on one wing half are located at the four flap support and drive stations, the two pylons, the fuselage side, and the aileron hinge and actuation stations. Unlike a multirib wing, there are no general cover support stations, this function being performed by the spanwise K-stringers. However, like a multirib wing, the truss plate stations may be thought of as trussed ribs distributed over short distances. The bulkheads shown at Station 91 (fuselage side) and the inboard pylon are principally tank barriers although they will also aid shear redistribution at those stations. The bulkheads shown at centerline (Station 0) and Station 61 are provided for cover support and fuselage bending load path, since the trussed-rib arrangement is not used in the center wing area. The isogrid design was chosen for the fuel and center wing bulkheads rather than sandwich panels in order to reduce fabrication cost and to avoid the use of core materials immersed in the fuel area.

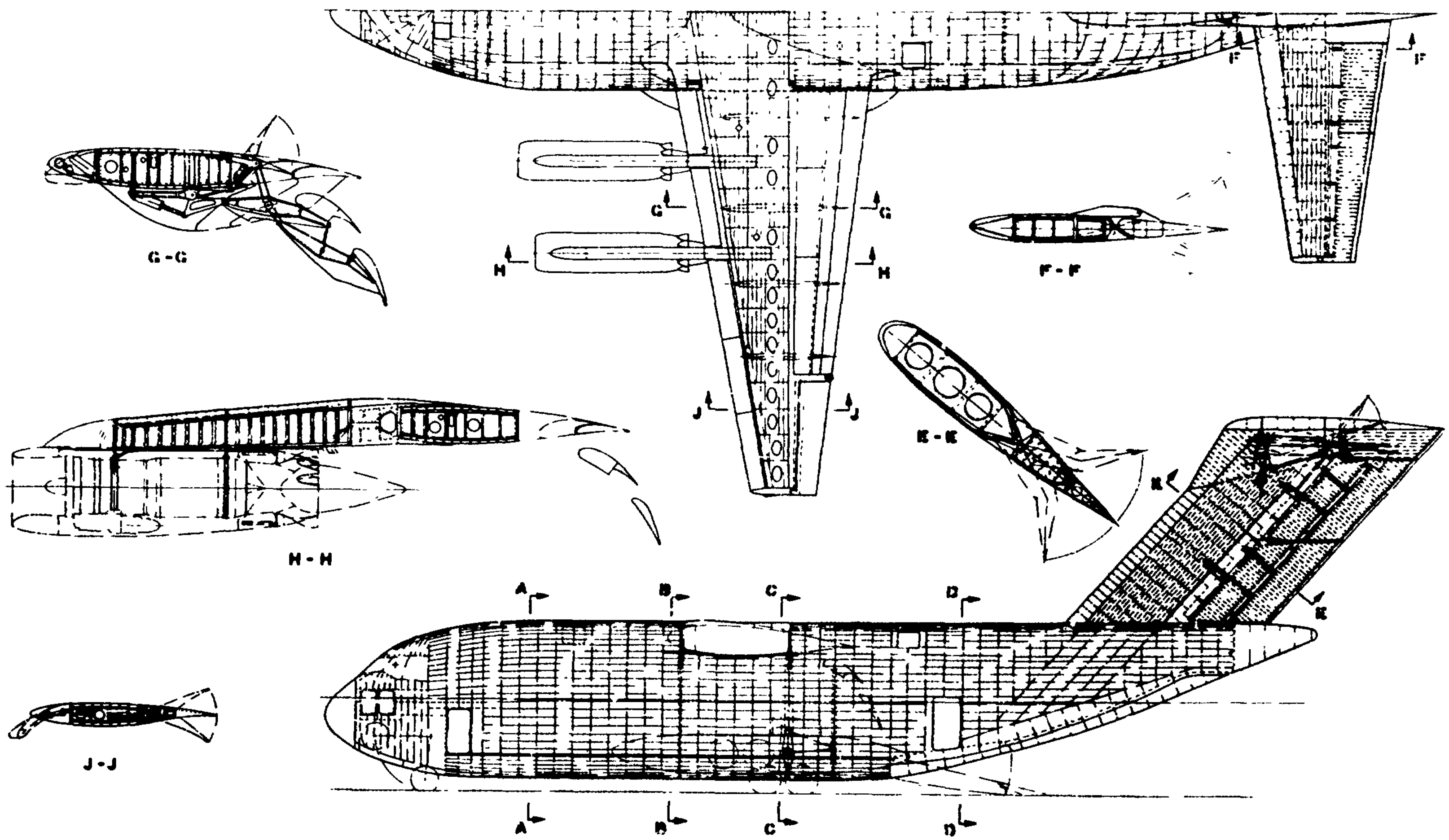


FIGURE 16. STRUCTURAL ARRANGEMENT – BASELINE METAL AMST (SHEET 1 OF 2)

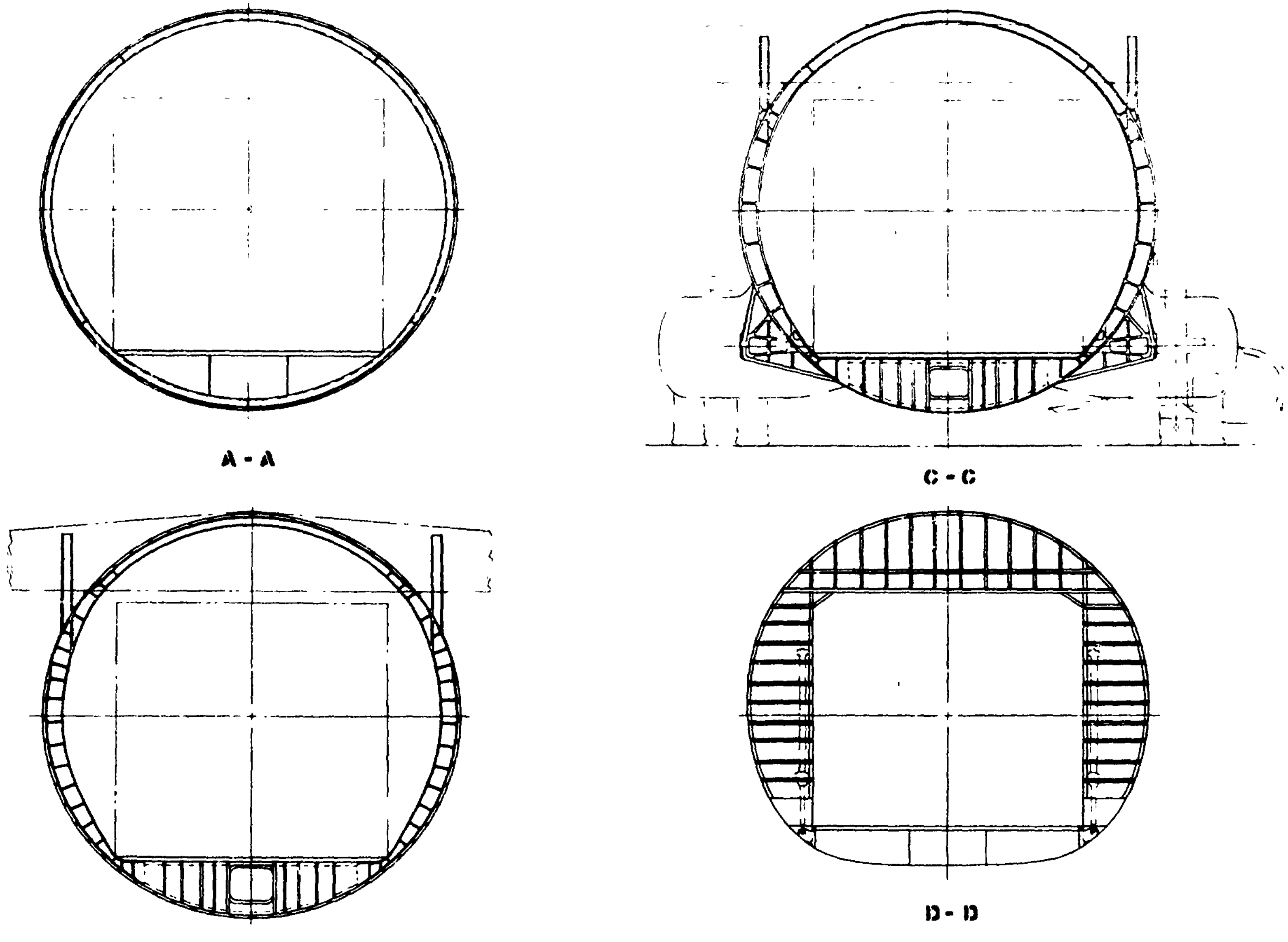


FIGURE 16. STRUCTURAL ARRANGEMENT – BASELINE METAL AMST (SHEET 2 OF 2)

The front and rear shear webs are solid-laminate construction because of the many systems that attach to these webs. The derby hat was the lowest cost and most efficient solid stiffening (Table 7). Although sandwich webs were included in the selected concept evaluated earlier (Table 5, Configuration C) the second lowest cost configuration utilized solid-laminate shear webs. Sandwich webs in this application appeared heavier and more costly in the practical case, wich required cutouts, potting, and buildups.

The wing utilizes high-aspect-ratio sandwich panel covers to avoid the cost and weight of rib supports necessary if spanwise stiffeners were used. The aluminum core is tapered to reduce penalty of sandwich covers on wing fuel volume. EDM machining is used to reduce core tapering costs.

The design utilizes spanwise stress relief strips for bolting. Glass plies are substituted for graphite plies along truss crests only. Ninety-degree and 45-degree layups are not interrupted at truss crests. The true "zero-degree layup direction" crosses truss crests in the tapered wing geometry except along the crest to which it is parallel; therefore a midchord crest is selected to define the basic zero-degree direction for two adjacent panels and local zero-degree directions are defined for the remaining panel and the two stress relief strips along the front and rear edges of the cover. See View S-S in Figure C-1 of Appendix C. Tape edges are progressively slit during automatic layup to accommodate areas of basic and local zero-degree convergence.

To reduce layup costs, the wing box graphite material was taken to be 0.010 inch thick per ply layer. The thicknesses required for the stiffness critical wing covers and webs are obtained by using balanced ply patterns of the 0/±45/90 layup system. The thicker plies have a negligible effect on laminate strength but would produce a slight reduction in bearing strength at bolted joints.

Although not pictured on Figure C-1, the centerline splice for wing covers and webs is considered to be an all-graphite (no titanium inserts), multi-fastener, double plate butt splice. The splice plates on upper and lower inside surfaces are tees, with the upstanding leg used for connection to the centerline bulkhead. The upper splice tee and external splice plate also carry a proportion of fuselage bending loads as well as acting as wing cover splices. This is suggested in Section W-W of Figure C-2. This major joint, as well as other major joints, would require additional detail design effort for proper definition.

Other major structural interfaces on the wing are at engine pylons and flap supports. Sections B-B and C-C of Figure C-1, as well as Sections C-C and D-D of Figure C-5, suggest these areas. The pylon horizontal reactions are taken into the wing only at the truss crests and are not distributed into the covers directly. To aid bolt bearing and load redistribution at these pylon load locations, stepped-lap titanium inserts are bonded into the outer cover chordwise doublers. Pylon load introduction fittings which span significant distances in contact with the box would be of titanium to minimize thermal mismatch.

The flap upper reaction point structure design depicted in Section B-B of Figure C-1 is conceptual in nature and was supplied to provide a basis for cost and weight estimation. Further design effort would be required to define detail strength design for this high load input area.

4.1.2 Fuselage

Figure C-2 of Appendix C illustrates the composite fuselage concept in the form of the MCE drawing produced for cost- and weight-estimating purposes. The basic isogrid concept selected for the main barrel which extends from Station 439 to Station 947 (42.33 feet) is depicted in View E of Figure C-2.

Figure 17 shows further detail of the fiber intersection scheme on which the basic isogrid concept is based. Fibers run continuously through the joint areas in three alternating directions during automatic layup of the grid, producing fiber-interlocked joints. While wrapping a single grid rib with three stacked bands at once, banding control separates the bands approaching each joint area to produce a wider single-ply layer at the intersection, thus avoiding fiber stackup. No holes are allowed through the grid intersection area, as in metal isogrid, thus avoiding severe stress risers from cut fibers at holes. An alternate layup scheme, also shown on Figure 17, offsets bands so that only two directions intersect at once, thus producing resin-rich grid ribs for optimum joint resin content. This alternate concept allows spacing of bands far enough apart, without apparent detriment to the structure, to allow holes at the intersections if desired for universal equipment attachment locations. The flared design does not add significantly to weight or facilities and layup cost, and was judged to be the safest conceptual approach. The offset joint is considered feasible both structurally and from the manufacturing standpoint and is under active consideration pending further test and manufacturing development effort.

The unidirectional material in the grid ribs is utilized for stiffness rather than strength. The grid primarily provides general shell stability and load redistribution paths, except at circumferential joints where it is the primary load path.

A secondary isogrid is utilized (on a 2-inch module within the basic 4-inch triangular module) to stabilize the triangular skin panels of the primary grid, allowing a significant reduction in unstabilized skin weight. No buckling of fuselage skin is allowed since the grid-to-skin joint would be sensitive to peeling forces generated by skin wrinkles. The secondary grid is accomplished easily by a change during the automated wrapping sequence from the 4-inch to the 2-inch module.

The grid dimensions and basic skin thickness were established for a maximum load area and for general stability. Lower load areas are accommodated by reduction in skin gauge only. Local prefabricated doublers may be inserted between the grid and the external skin plies for load introduction and cutouts. The secondary grid may be filled at joint areas, such as indicated in Section J-J of Figure C-2, to allow insertion of internal reinforcements. The basic grid dimensions, however, are uniform throughout a barrel section to accommodate the wrapping process.

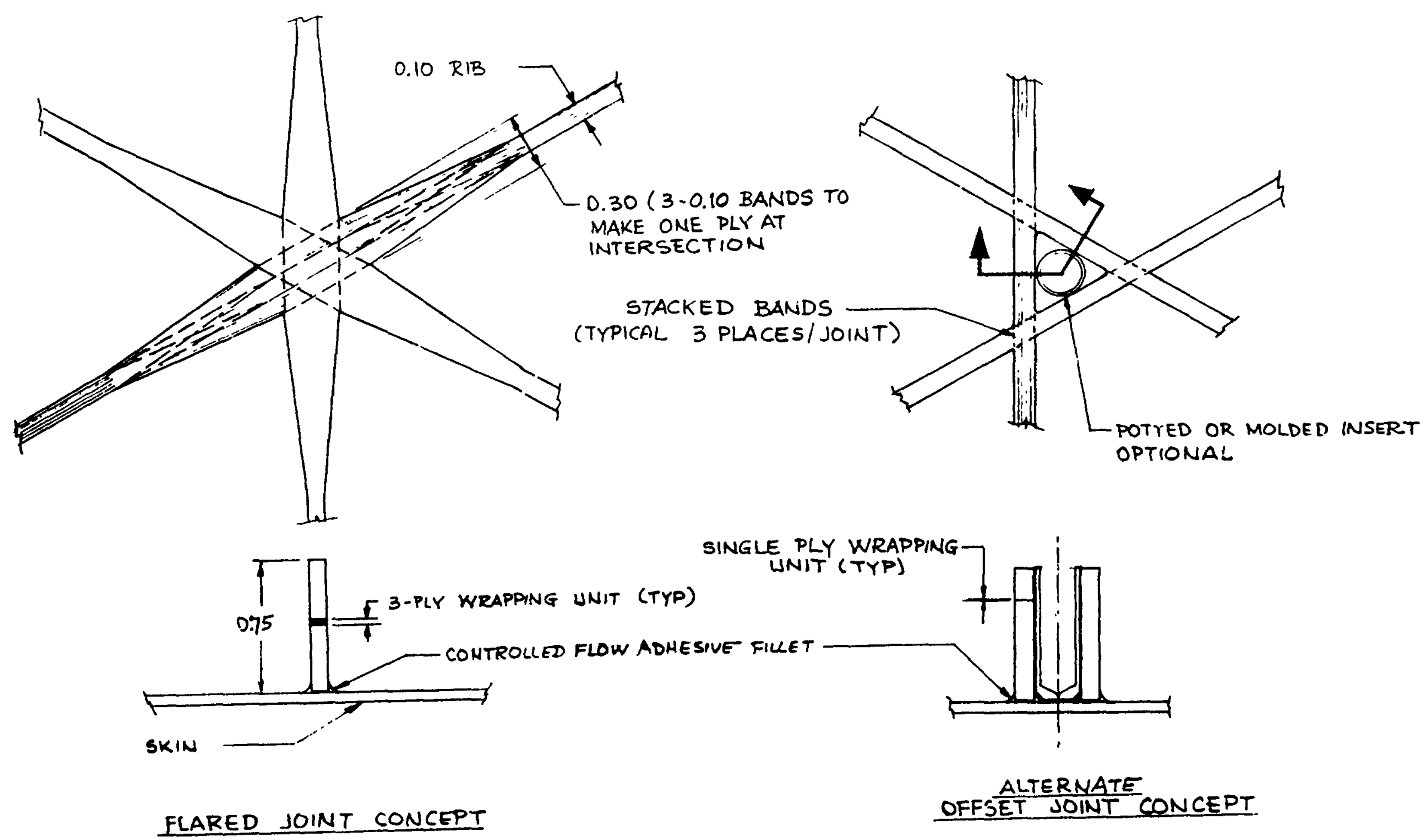


FIGURE 17. DETAIL – ISOGRID JOINTS

A feature of the "turnaround" joints at each end of each fuselage barrel is the use of stress concentration relief for the bolt holes through the end V-joints. Midway through the wrapping procedure, tapered fiberglass inserts are added to the layup and wrapping is then continued. The bolt holes are in the glass wedges rather than in unidirectional graphite.

The aft fuselage section (Station 982 to Station 1437) is also of composite isogrid design. The tapering section is fabricated by dropping circumferential subdivisions and reducing grid rib height in several stages in the aft direction. The procedure is further detailed in Paragraph 4.3.3, Composite Fuselage Manufacturing and Assembly Outline.

In the open aft cargo loading section, an inner shell or space framework is constructed of pultruded composite tubing to form a large-scale open isogrid (space frame) to stiffen the aft fuselage in torsion. The inner and outer shells are connected by solid-laminate deep frames.

The fuselage nose section is of thin graphite-faced sandwich panel and glass or Kevlar-49 fabric frame construction because of severe contours in that region. Graphite caps are included in the frame buildup. The three fuselage sections are bolted together with tension bolts at circumferential joints. The cargo and ramp flooring are made of boron-infiltrated extruded aluminum planking. Metal fittings are used in all critical joint areas. The floor support structure is an egg-crate arrangement of derby-hat-stiffened solid-laminate panels.

4.1.3 Empennage

Figure C-3 of Appendix C shows the MCE drawing for the vertical tail box and associated rudder structure. The remainder of the vertical tail is conventional metallic design. The tip is a fiberglass antenna cover. The box is a sandwich panel multirib construction (except for front and rear spar webs which are derby-hat-stiffened solid laminate) with three full-span spars. Toward the root area, the five pultruded spar caps are utilized to collect and transfer bending load into the fuselage. The vertical box also carries the pivot fitting for support of the horizontal stabilizer. This structure is undefined on Figure C-3 but would be of forged and machined aluminum identical to the metal baseline.

The forward and aft rudder structures were adapted from another program (Reference 6) and have been described earlier in Paragraph 2.4.4.

The horizontal stabilizer box design has also been described by reference to the similar wing box design. See Figure C-4. It differs from the wing box because of reduced interface requirements and no fuel containment. The truss plate stations occur at each elevator hinge and actuator. A single wide truss plate area serves for stabilizer pivot fitting load introduction (at Station 14.20), and stabilizer actuation (at Station 6.00). The sandwich covers taper to solid laminates for splicing at the centerline. See Section E-E, Figure C-4. Aluminum bathtub fittings serve to splice right- and left-hand graphite K-section stringers across the centerline bulkhead.

The ply thicknesses used for horizontal stabilizer box covers and webs are 0.0055 inch per ply, unlike the wing covers which were 0.010 inch per ply, since the stabilizer gauges and overall sizes are approximately half those of the wing.

4.1.4 Interfaces

Figure C-5 is the composite airplane structural scope drawing produced to show interface relationships among structural components and to show overall assembly. Reference to this drawing and the wing drawing (Figure C-1) has already been made for pylon/wing and flap/wing box interfaces.

The wing/fuselage interface is portrayed in Figure C-5 (scope) and also in Figure C-2 (fuselage). Reference has been made to the fuselage bending load paths that must be provided through the wing center section (e.g., at Wing Station 61, right- and left-hand, and at fuselage centerline/wing centerline splice). See Figure C-5, Sections F-F and G-G. The wing vertical reactions are taken into the fuselage frames at Stations 703 and 847 via shear panels that attach to wing front and rear shear webs, Figure C-5, Sections H-H, J-J, and L-L. This scheme was retained from the metal baseline since it effectively isolates wing bending from the fuselage, avoiding adverse deformations in the fuselage side and floor during wing flexing. Pitching moment loads from wing torque are also reduced by reacting the wing into frames which are farther apart than the wing box chord distance at Station 91.

Wing horizontal reactions are taken from the wing lower cover into the fuselage by the wing attach tee. The isogrid fuel bulkhead aids shear transfer from upper to lower wing surface. This wing attach tee becomes an angle forward and aft of the wing box and is exterior to the fuselage contour, but under the fairing. The attach tee is thick adhesive bonded to the wing lower cover and bolted through the truss crests. It has a flexible thinner section for mechanical ties to the fuselage side directly below the wing, and the fore and aft extensions are also mechanically fastened to the fuselage side for possible removal with the wing.

The main landing gear interface with the fuselage is essentially the same as for the metal baseline since the large forged aluminum frame is retained except for local graphite sections. The nose gear attaches to aluminum spider fittings that bolt to the graphite wheel beams.

The vertical tail attachment to the fuselage was changed from the metal fuselage scheme because of the nature of the isogrid shell. In the metal fuselage, the vertical tail spar caps penetrate the fuselage pressure shell to provide an efficient shear tie to slant bulkheads, which in turn distribute the V-tail torque load into the skin and stringer shell. This load must be distributed over a significant distance as evidenced by the extension of the slant bulkheads nearly to the bottom of the fuselage, Figure 16, in order not to locally overload the shell. The isogrid shell, on the other hand, has the capability of more quickly dissipating concentrated loads because of grid redundancy. In the composite fuselage, therefore, the spar cap loads are transferred to large fittings interposed between the V-tail box and the fuselage. These fittings are bolted through the fuselage skin to corresponding load distribution fittings inside the isogrid shell. The fittings tie to vertical frames which accommodate the inner and outer isogrid vertical geometry (as slant bulkheads would not). The separation of swept tail loads into vertical and horizontal components requires the resulting horizontal kick loads to distribute into the outer shell. Local longitudinal intercostals may be required to dissipate this load into the shell. The detail design of the entire aft fuselage requires a finite element analysis for adequate definition of internal load distribution; however, the scheme is efficient and sufficiently redundant, especially as to the inner isogrid space frame, that further investigation should find this arrangement conservative as to weight and stiffness.

Additional interfaces occur at each place where retained metal structure must connect to graphite structure and thermal imbalance must be accounted for. Notable areas are the fixed and movable aluminum leading edges to graphite composite boxes, cargo floor to fuselage shell, and main fuselage frames to fuselage shell. Program scope did not allow detail assessment of these problem areas; however, initial assessment indicates that magnitude of thermally induced loads shared between two such continuously and rigidly connected parts is dependent upon relative areas and stiffnesses (EA's). The compatibility-induced strain is shared, producing a running stress level within each part and a sharp load increase at the very ends of the part due to the sudden discontinuity. Both loads are calculable and the problem area is located at the part ends. The magnitude of this end load which must be resisted, if the parts are to stay together, is larger for larger cross sections. For small parts, the thermal mismatch end load is amenable to design solutions; the larger joined dissimilar components must be studied for feasibility of being joined by direct connection. Redesign to provide thermally flexible load paths, as in certain areas of SST design, is a distinct possibility for joining large aluminum to graphite components. In the weights associated with the present designs, there is no allowance for thermal incompatibility.

4.2 STRUCTURAL ANALYSIS

The analysis performed in this program was based upon the detailed finite-element analysis conducted on the metal prototype YC-15 aircraft. Overall distributions of the external loads (as shear, bending moments, and torques) were compiled for each part of the structure. Precise definition of the internal loads was defined at each of four stations on the wing, fuselage, and horizontal and vertical tails. These data were then scaled appropriately for the initial resized composite aircraft (Figure 3), which had an 8.4-percent decrease in wing area with compatible decreases in the empennage areas. The fuselage was not resized because of payload considerations. These distributed loads (scaled where necessary) formed the basis of the overall sizing of the composite structure skins and members. In addition to this, the specific concentrated loads at wing-to-fuselage interface, pylons to wing, flap support bracket loads, etc., have been estimated for the resized aircraft on the basis of the metal YC-15 analysis.

The analysis served to support the preparation of cost-estimating drawings and can be considered to relate to Airplane B of the later payoff studies, since the initial composite and the completely resized aircraft (B) differ by only 0.69 percent in wing area and 0.80 percent in gross weight.

The load data generated were intended to serve as a reference level to establish the amount of composite material necessary to resist the applied load. However, the ratio of specific strength to specific stiffness changes significantly in the process of replacing a metal design with composite materials. Consequently, it was found that a wing design based on strength considerations alone was seriously deficient with respect to stiffness and flutter requirements.

4.2.1 Wing and Empennage Boxes

Sizing for the wing and empennage was re-established on the basis of suitably scaled stiffness distributions relative to the metal design. Specifically for the resized composite wing, it was shown to be necessary that the bending stiffness (EI) and the torsional stiffness (GJ) were 0.79 of the metal values from a stiffness standpoint, only 0.62 as high for bending strength considerations, and 0.75 as great for torsional strength. Adherence to these stiffness requirements would have ensured the same flutter margin as exists on the metal aircraft. However, doing so made the composite wing excessively heavy, so the weight was subsequently reduced by a flutter sensitivity analysis to redistribute a smaller amount of material in a more optimum manner. Budgetary and schedule requirements did not permit either a flutter analysis or an in-depth static strength analysis of the eventual designs so established. The low emphasis on analysis of the designs is in keeping with the major emphasis on cost factors.

In the case of the horizontal stabilizer, it was found that the respective EI and GJ ratios for composite with respect to metal were 0.62 (EI) and 0.76 (GJ) based on strength and only 0.62 based on stiffness considerations. In other words, the horizontal stabilizer appeared to be strength critical. The vertical stabilizer, likewise, seemed to be strength critical, with the resized aircraft needing 0.62 times as much EI and 0.83 times as much GJ as the metal aircraft when viewed from a strength standpoint and only 0.72 times as much material from a stiffness point of view. Actually, because much of the material needed to establish the EI for the metal baseline YC-15 stabilizers also contributed simultaneously to an excess in GJ, the composite vertical and horizontal stabilizers were both found to be stiffness critical when strength checks were made on the initial design. This was because they matched metal stabilizer stiffness on an equivalent bending and torsional frequency basis. Subsequently, material was removed from the designs of all the primary box structures based on reducing the estimated flutter margin of the metal YC-15. Approximately 40 percent was removed from the wing skins. Since all these structures are critical in stiffness and not in strength, and no definitive flutter analysis or static aeroelastic assessment was performed, it is not appropriate to provide strength margins for the composite design. A survey of ultimate static strengths showed no negative margins arising from the decrease below the equivalent structure.

4.2.2 Fuselage

In contrast with the situation for the wing, much of the composite fuselage design was governed by stability or strength considerations. Starting with the aft fuselage at the front of the opening for the main cargo door, and progressing forward, the internal load distribution at station 982 was assessed to be equivalent to a general maximum compressive load intensity of 4000 pounds per inch ultimate, with a local peak value of 7000 pounds per inch being handled by local reinforcement. The torsional shear flow given by the computed load and area between the outer skin and inner open truss was determined to be 833 pounds per inch. Ultimate in-plane strength checks were satisfied. The buckling of the isogrid fuselage skin was then checked for these combined loads by replacing the stiffened skin by its equivalent isotropic panel. The open truss struts around the aft cargo door opening were sized by the same loads, with checks on Euler column stability and strength. At this station, as well as at the outer three stations analyzed and over the entire fuselage, the initial design was adjusted in the light of the computations. Moving forward to the rear spar and main landing gear attachment at Station 847, the skin was checked for ultimate strength, skin buckling, general instability, and rib crippling for the computed load intensity of 2250 pounds per inch ultimate in compression and an allowance for a small shear load. The small stiffeners within each basic isogrid triangle were found to be necessary to stabilize the skin against buckling which could not be permitted because it would induce delaminations between the skin and stiffener. At the forward spar attachment, Station 703, the design load condition is 1570 pounds per inch ultimate and the same four checks were made. The analysis plan called for a check at Station 439 but the barrel winding manufacturing scheme has the fuselage cross section maintained constant back to Station 617, where the loads are higher, so the final check was made there. The purpose of the stress-analysis, for this project, was not an in-depth look at all the details. Rather, it was to establish the order of magnitude of the amount of material required for weight and cost purposes. A provision of 10 percent was made for undefined structure to deal with local stress concentrations and reinforcement. The major fuselage frames were not stressed but sized directly from the metal YC-15 design to provide equivalent strength and stiffness.

4.3 MANUFACTURING AND ASSEMBLY TECHNIQUE

The following descriptions of manufacturing sequences for the major conceptual composite structural components are for any of the three airplanes defined in the payoff studies, Section 6. Where overall dimensions of parts are stated, they refer to the fully resized aircraft (Airplane B) for which the raw estimates for manufacturing labor, tooling, and quality and reliability assurance were obtained.

4.3.1 Wing Box Fabrication and Assembly Outline

The wing box skins are approximately 634 inches long and 126 inches wide, tapering to 26 inches at the outboard tip. The wing box upper, lower inner, and outer graphite-epoxy skins that contain some fiberglass-epoxy stress relief strips are automatically tape laid, individually densified, and staged, Figure 18. The covers (including pultruded composite K-spar caps and K-sections for the W-truss webs) are assembled to aluminum honeycomb core which was electrically discharge machined (EDM) to contour, Figure 19. Figure 19 also illustrates the access panel door assemblies, the honeycomb core, and the densified inner and outer wing skins that are transferred

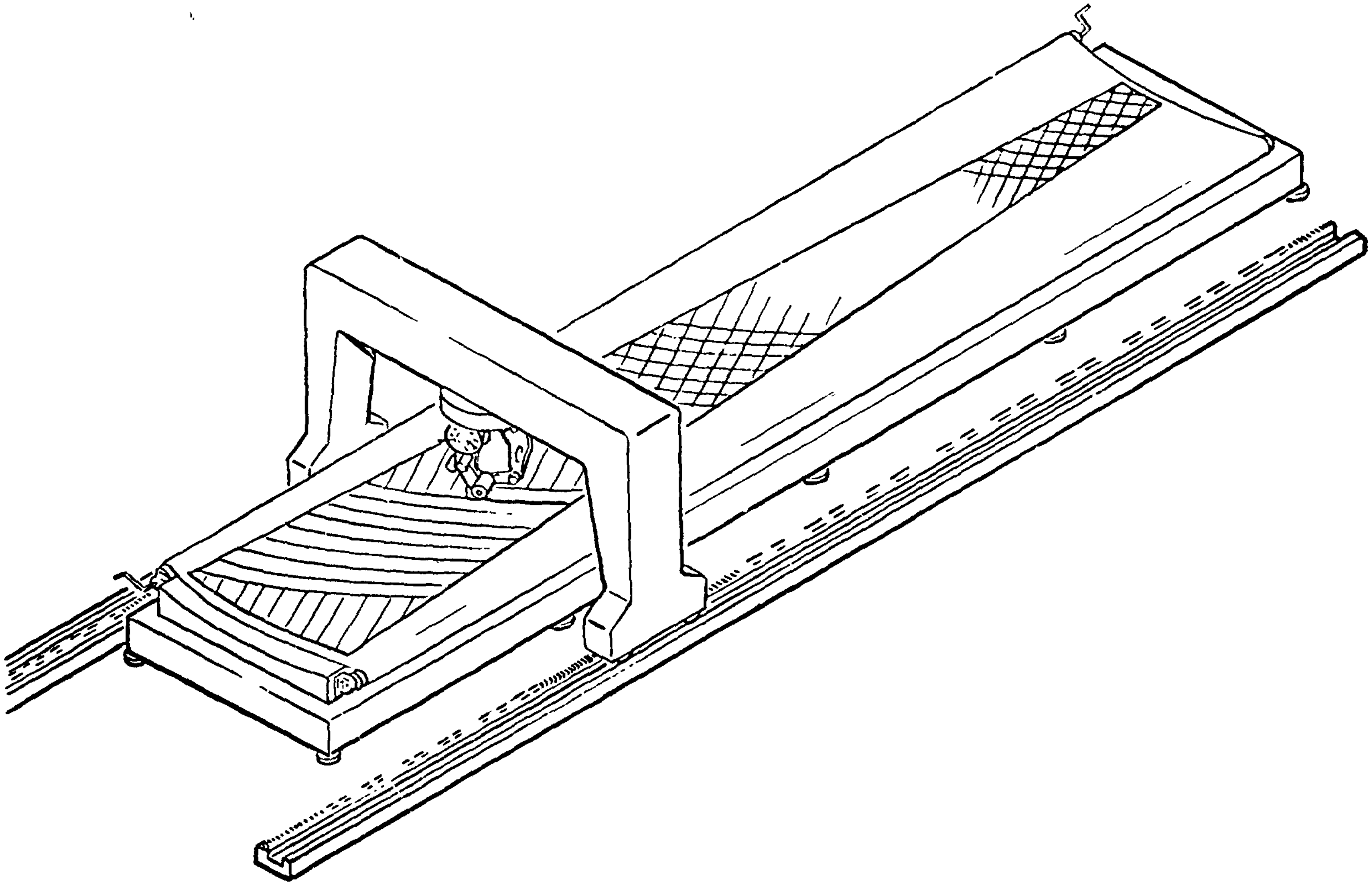


FIGURE 18. TAPE LAYUP OF GRAPHITE–EPOXY WING SKINS

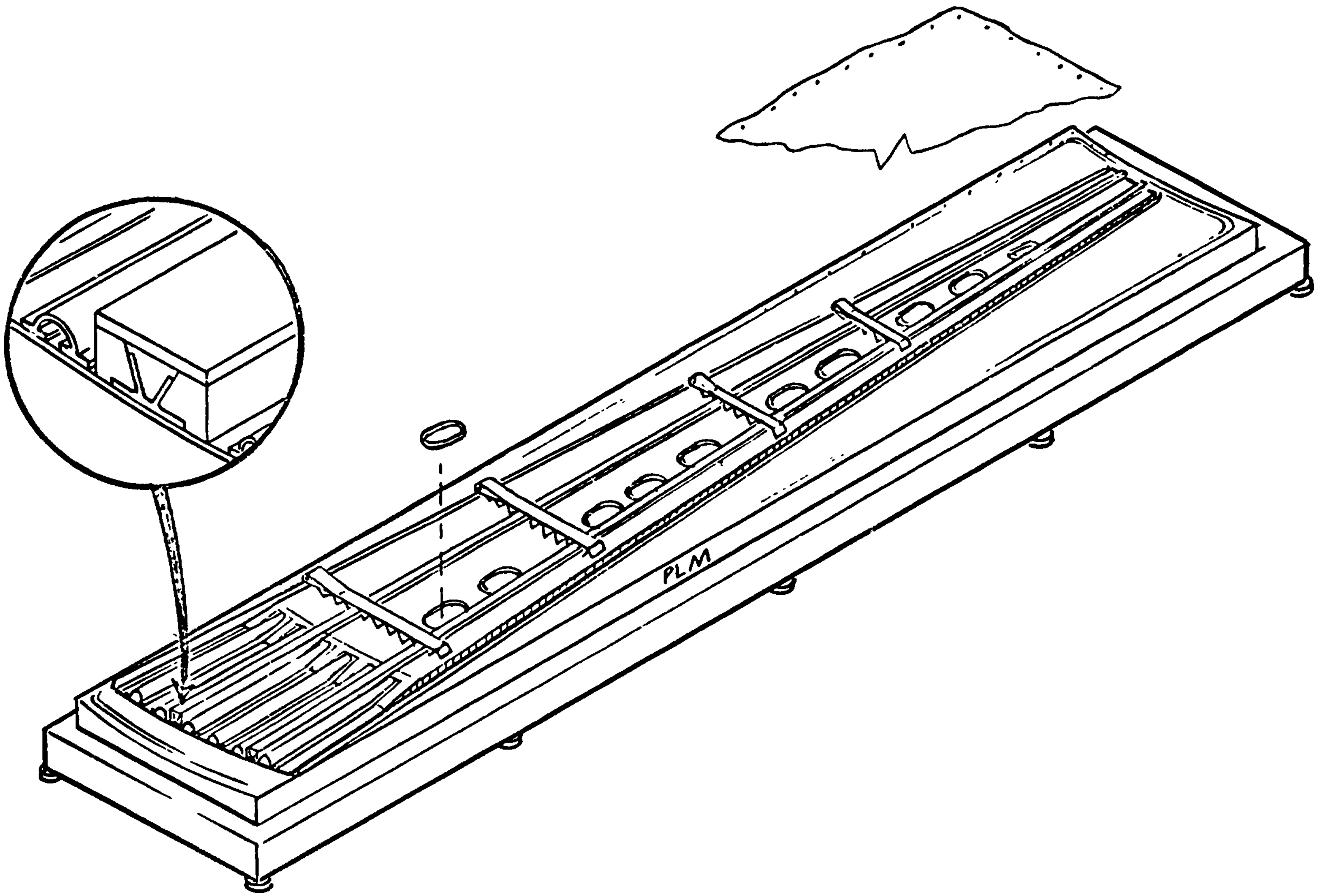


FIGURE 19. COCURING AND BONDING WING SKINS

to and assembled on the outer skin PLM. The wing box covers are then cocured and bonded in an autoclave at 350°F and 40-psig pressure for 4 hours.

The K-caps and K-section details are pultruded at the rate of one foot per minute through variable pultrusion dies, Figure 20. Ply orientations include 0-, $\pm$45-, and 90-degree fibers. The number of plies being pultruded can be changed, Figure 21, to allow tapering of the composite through a variable pultrusion die. The pultrusions are placed in densifying and staging dies prior to cocuring and bonding to the wing box skins (Figure 20). Figure 22 shows similar pultrusion, staging, and transfer operations for the access door jamb assemblies.

The derby-stiffened, solid-laminate front and rear spars are cut, formed, and cured as shown in Figure 23. The isogrid bulkheads are tape wound and cured as shown in Figure 24. The spars and bulkheads are aligned and positioned on the wing box lower skin.

Irregular shaped attach angles are automatically tape laid in a flat plane, densified, staged, and then trimmed net as shown in Figure 25. The attach angles are heat formed and installed in their proper locations on the wing box upper skin after applying a layer of release film to the skin inner surface. This allows secondary thick adhesive bonding of the attach angles to the substructure and the removal of the upper skin for the next operation after the angles are formed to fit.

Hydroforming of the aluminum W-truss web beaded panels is accomplished on a Verson direct acting hydraulic press, Figure 26. The beaded panels are shown in an exploded view of the wing box, Figure 27.

Small composite W-truss webs and straps for reacting flap loads, Figure 28, are automatically tape laid and densified in the flat condition, heat formed, cocured, and cut into individual components. The completed small W-truss web/straps are secondarily thick adhesive bonded to the upper skin at the same time the upper skin is bonded to the wing box substructure.

Figure 29 shows the complete flow diagram for the fabrication and assembly of one wing box.

4.3.1.1 A Note on Tooling for Large Moldings

All PLM's, exterior molds, dies, and mandrels used to fabricate composite components will be designed to allow control of heatup rates. This is essential to obtain the maximum strength requirements of the composite/resin systems being used.

The wing box skins vary from a 0.20-inch-thick solid-laminate inboard to an aluminum honeycomb core sandwiched between two 0.070-inch-thick skins outboard. The wing box skin PLM's will be designed to allow forced heating from the inside, as well as autoclave exterior heating. This type of tool heating will be required to maintain heatup rates (°F/minute) for thick laminates being cocured and bonded at the same time as thin laminates. Similar systems of controlling heatup rates during the fabrication of fuselage sections will also be incorporated.

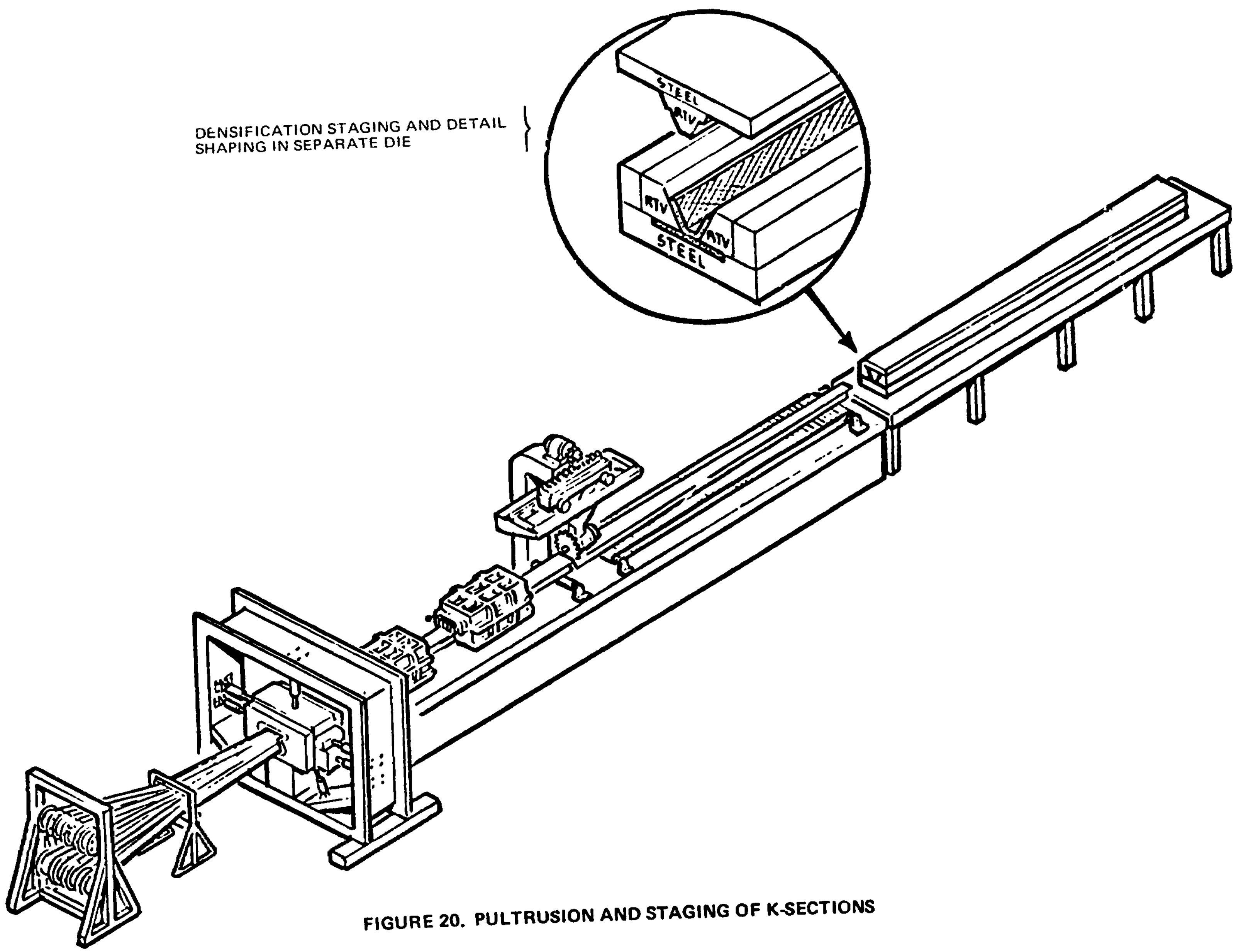


FIGURE 20. PULTRUSION AND STAGING OF K-SECTIONS

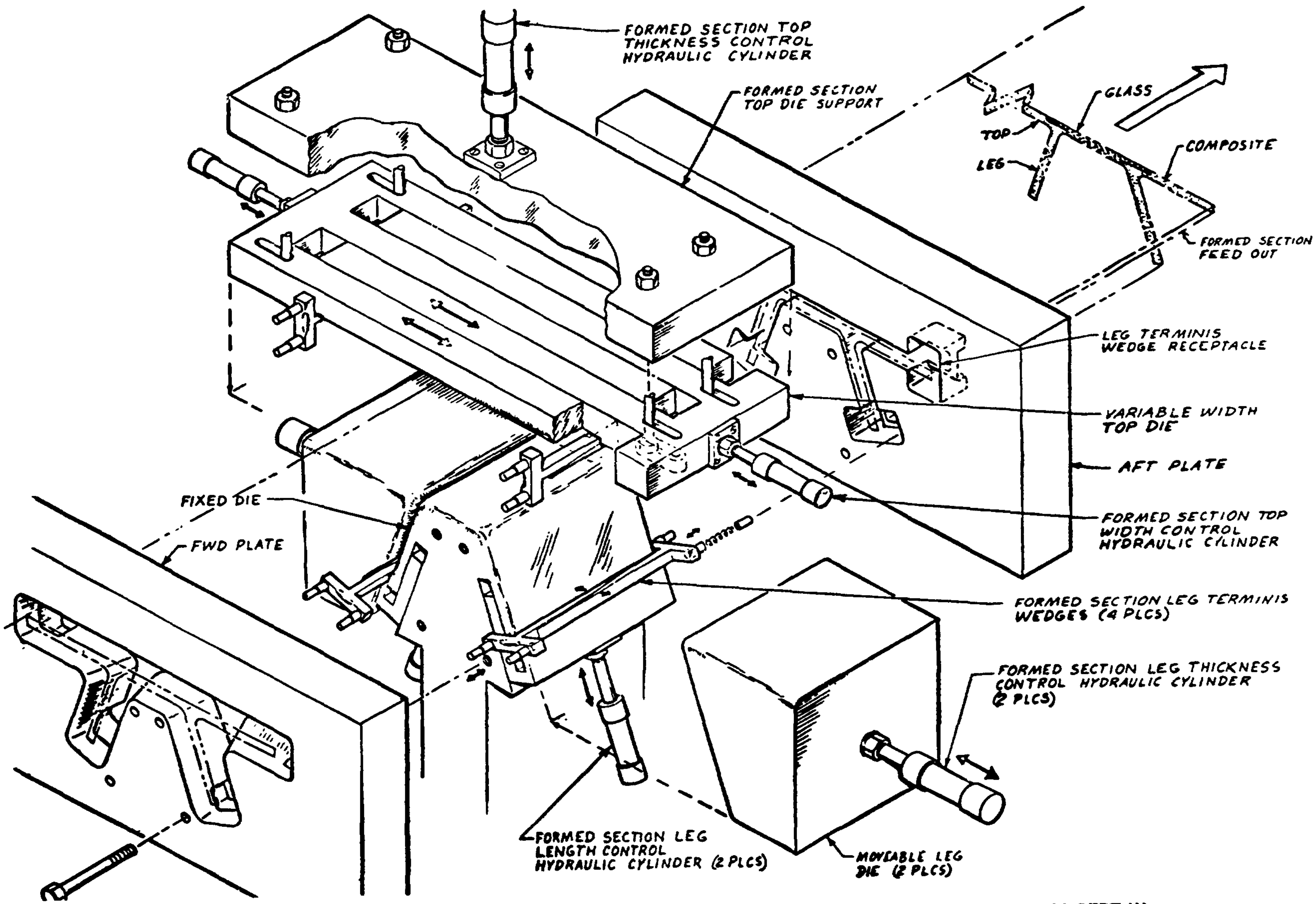


FIGURE 21. VARIABLE PULTRUSION DIE FOR K-SHAPED COMPOSITE FORMED SECTION – CONCEPT III

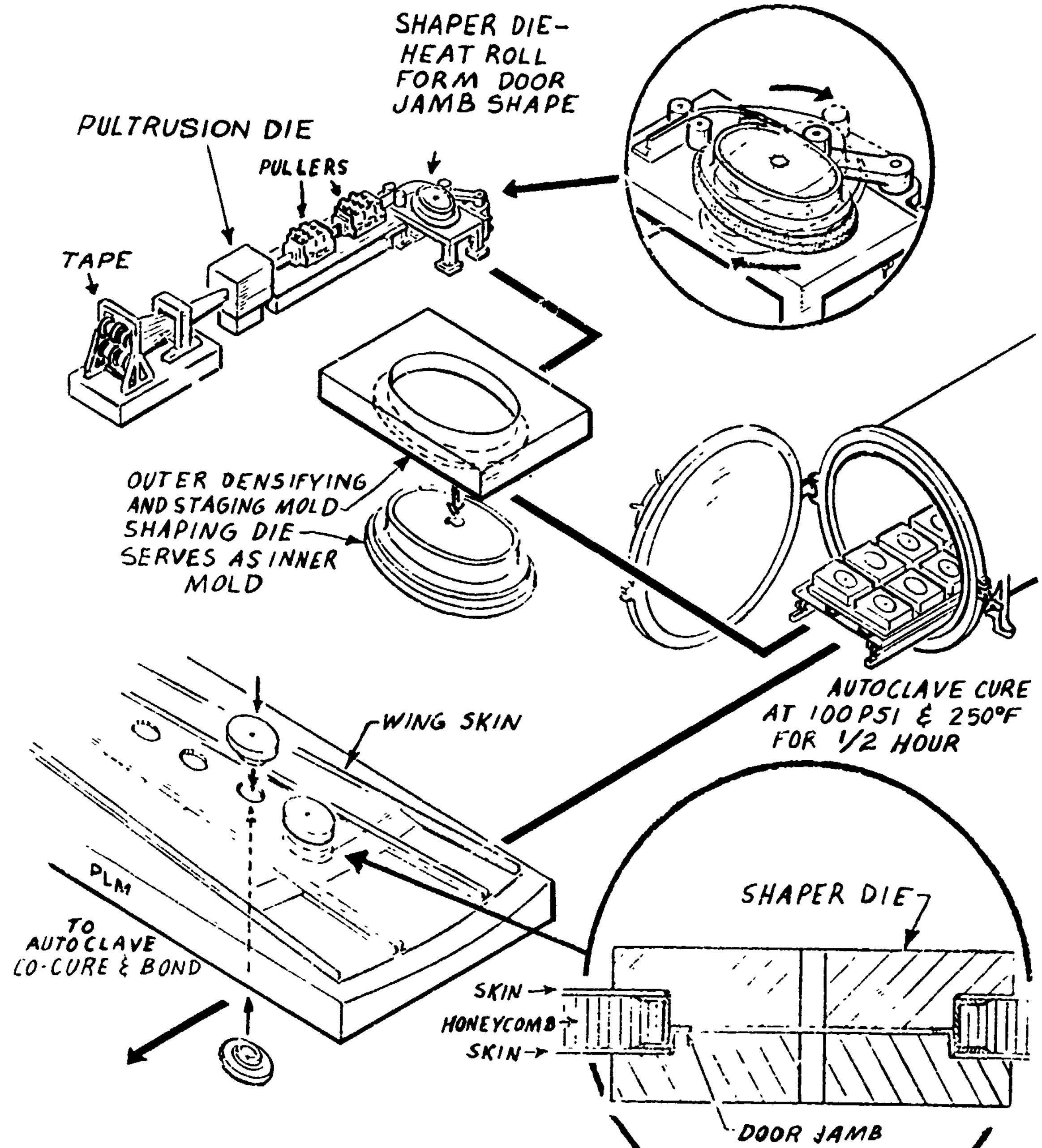
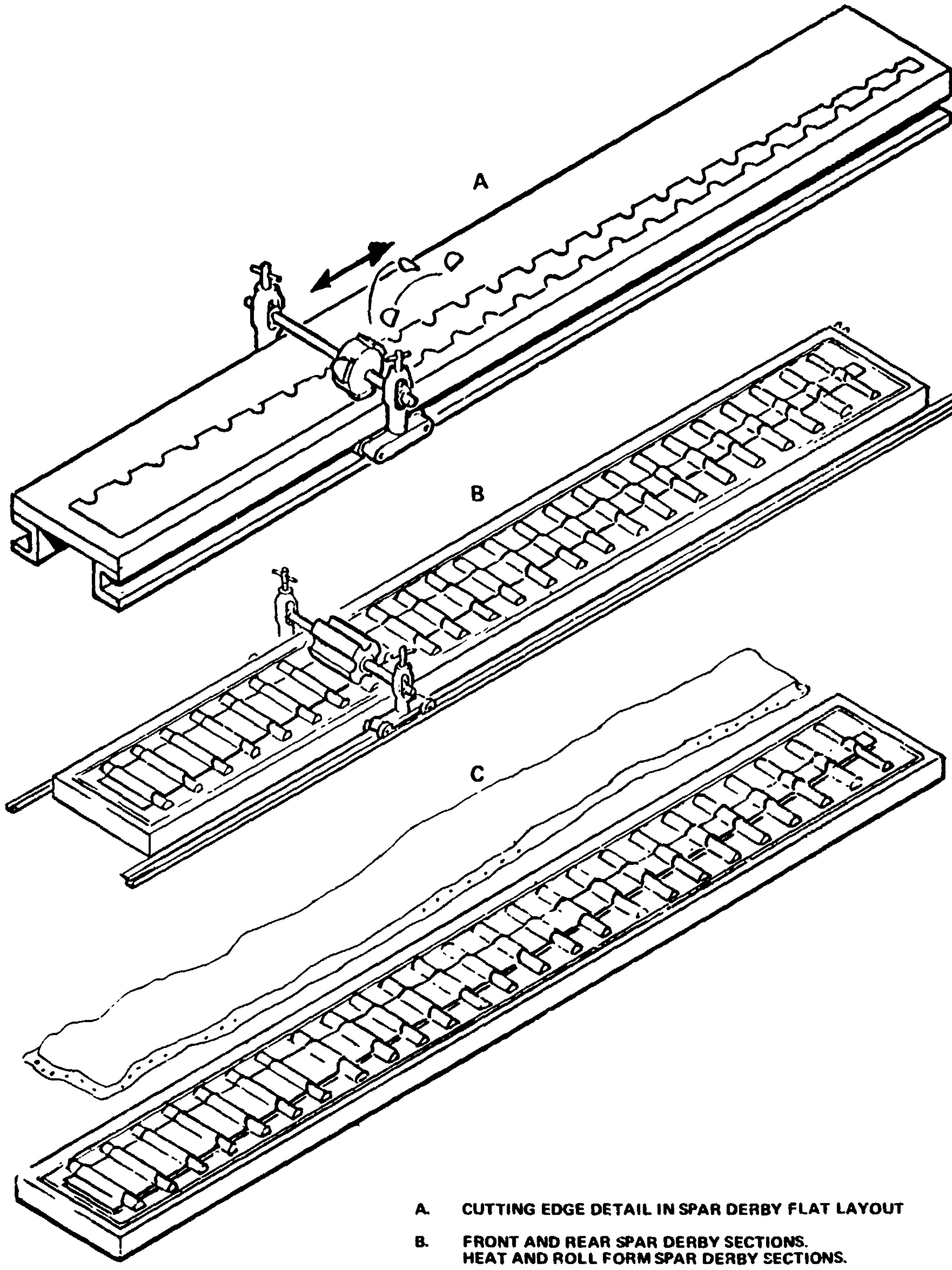


FIGURE 22. WING ACCESS DOOR JAMB DETAILS



A. CUTTING EDGE DETAIL IN SPAR DERBY FLAT LAYOUT

B. FRONT AND REAR SPAR DERBY SECTIONS.
HEAT AND ROLL FORM SPAR DERBY SECTIONS.

C. CURING FRONT AND REAR SPARS

FIGURE 23. FABRICATION OF WING BOX FRONT AND REAR SPAR WEBS

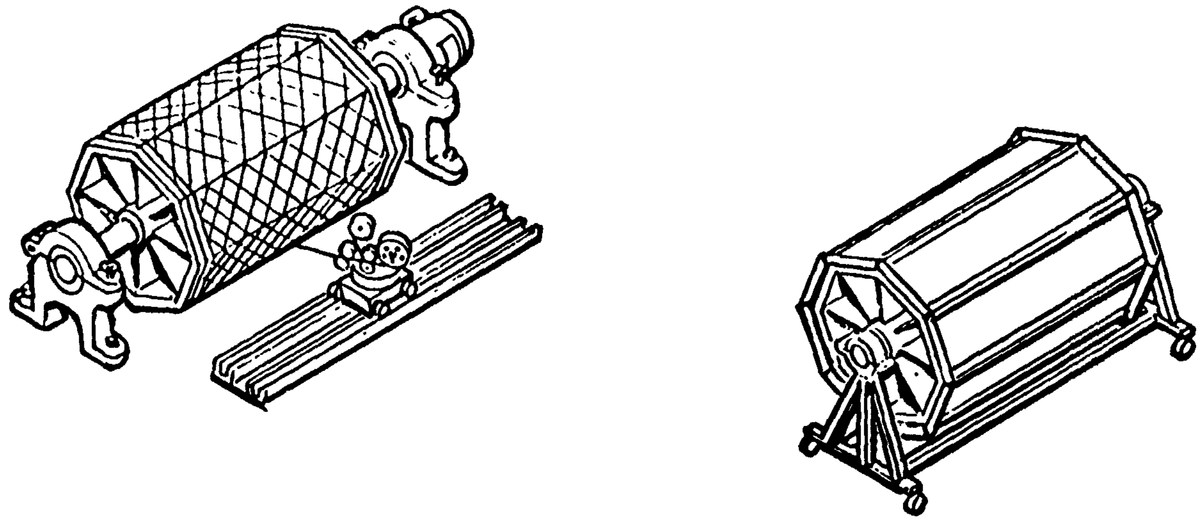


FIGURE 24. TAPE WINDING AND CURING ISOGRID BULKHEADS

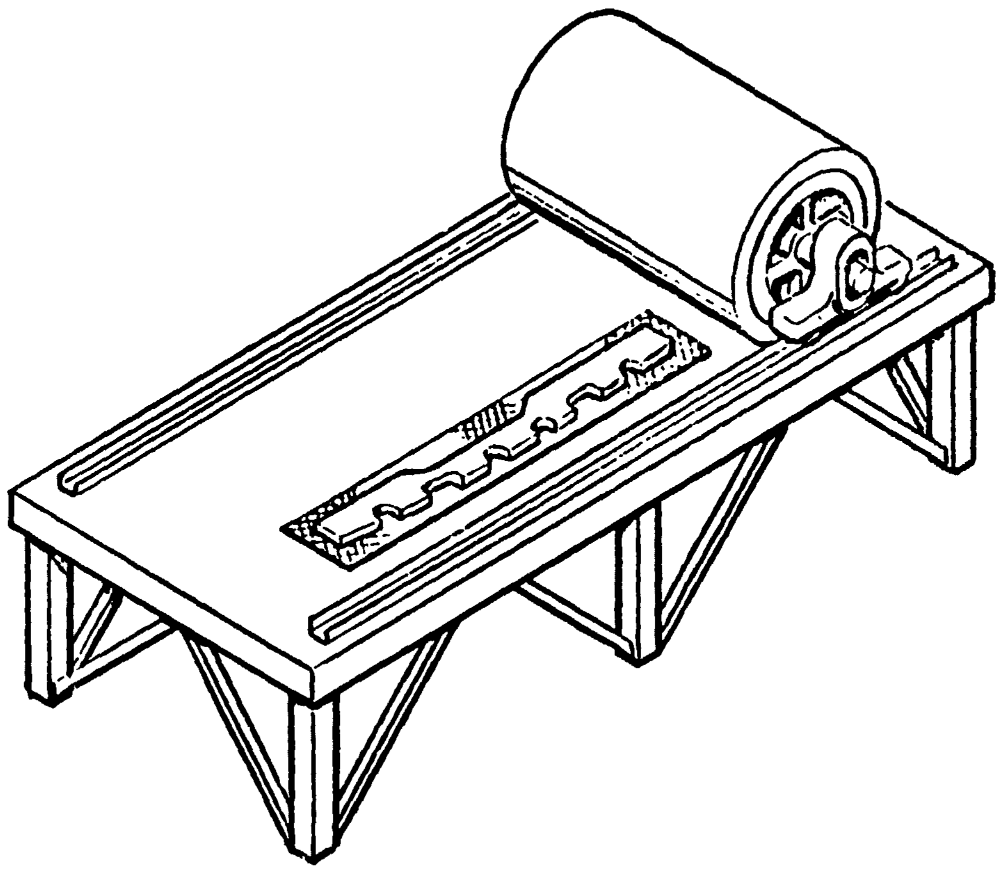


FIGURE 25. TRIMMING IRREGULAR CUTOUTS IN WING BOX ATTACH ANGLES

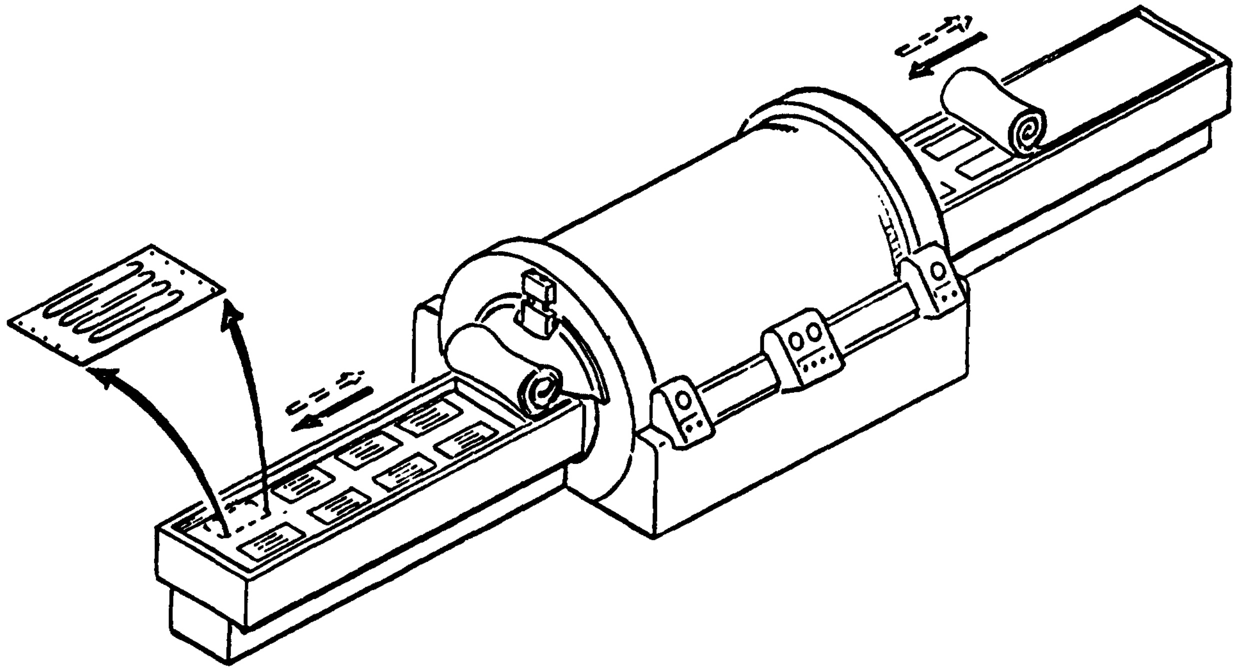


FIGURE 26. HYDROFORMING W-TRUSS WEB BEADED PANELS

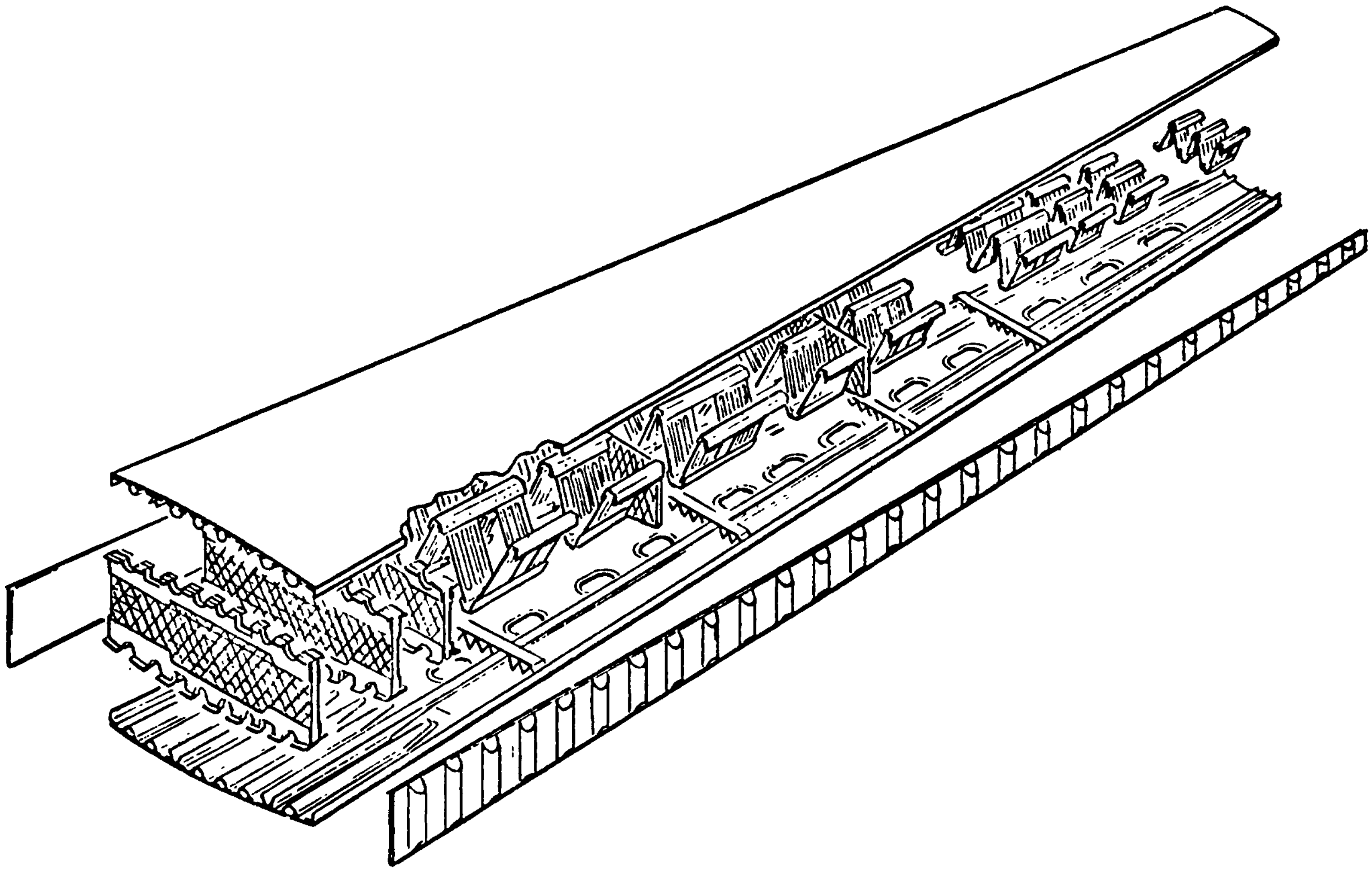


FIGURE 27. EXPLODED VIEW OF COMPOSITE WING BOX STRUCTURE

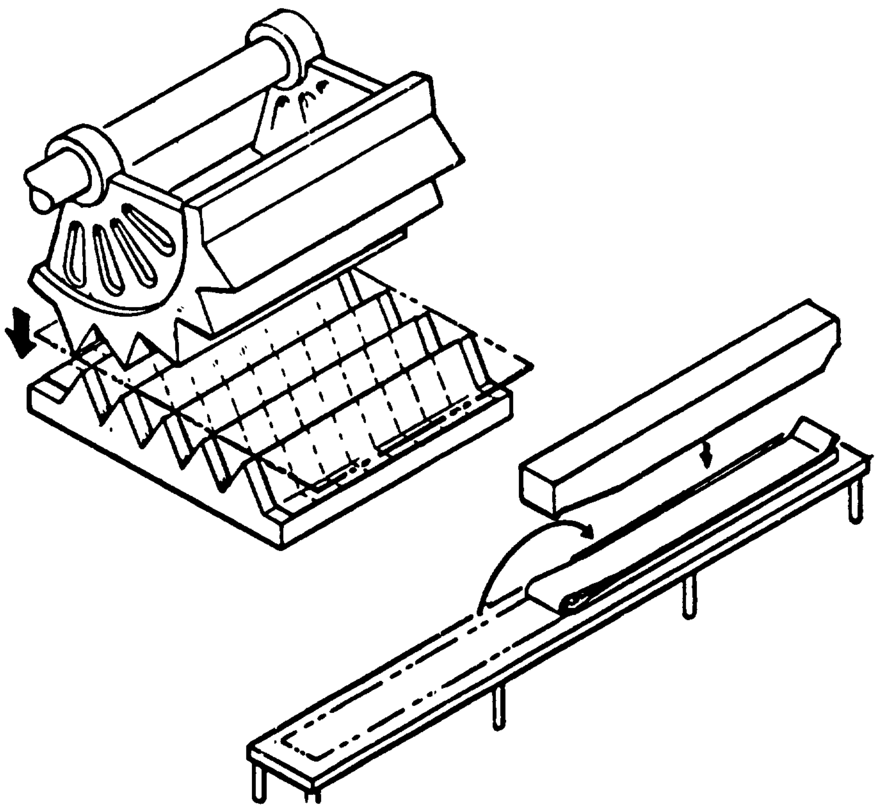


FIGURE 28. FORMING SMALL W-TRUSS/STRAP COMPOSITE DETAILS

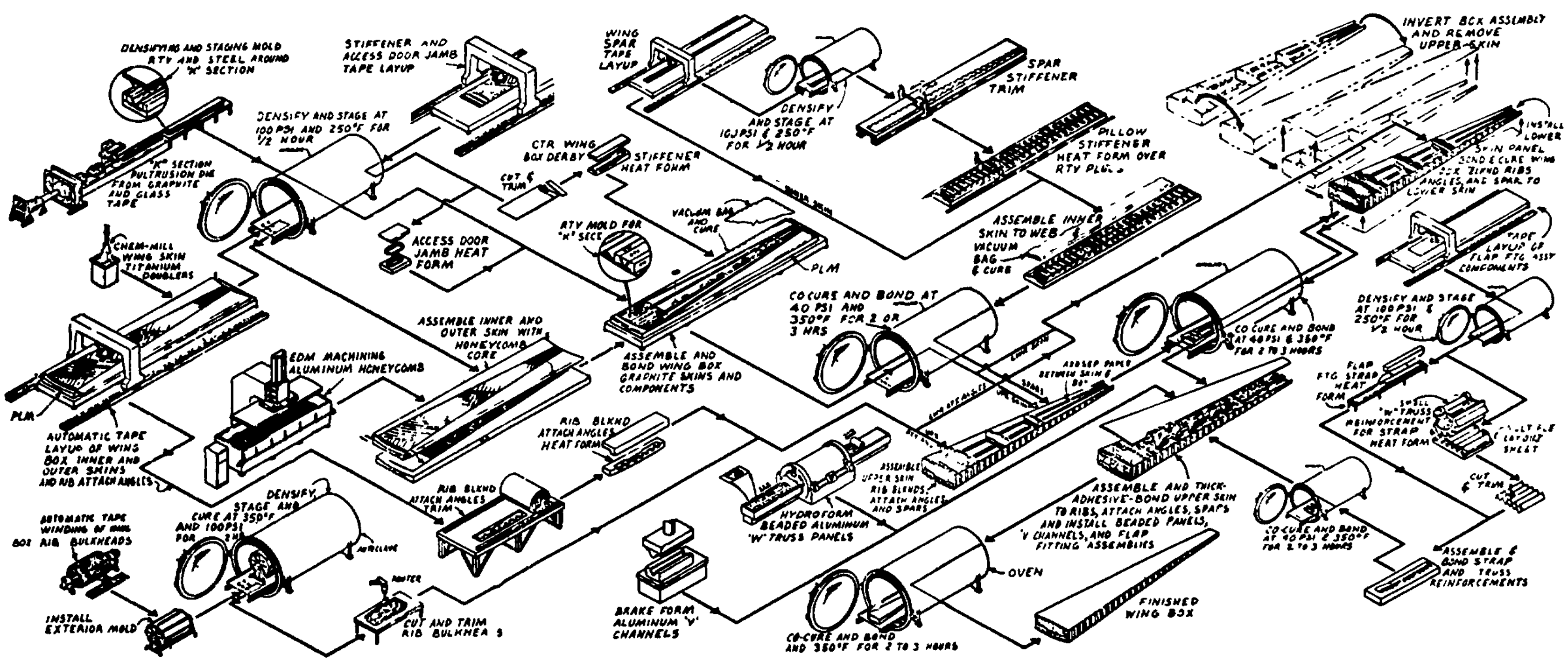


FIGURE 29. GRAPHITE-EPOXY WING BOX FABRICATION FLOW DIAGRAM

4.3.2 Horizontal and Vertical Stabilizer, and Rudder Manufacturing and Assembly Outline

The horizontal stabilizer is basically a W-truss web configuration very similar to the wing box except for size. The fabrication and assembly operations are very similar to those shown in Figures 18 through 29, with the following exceptions:

The access doors shown in Figures 20 and 22 are not required in the horizontal stabilizer, though access holes will be located in the front and rear spars of the vertical stabilizer for removing inflatable mandrels after cure. The same access holes will also be used for installation of metal details, such as the beaded web panels and hinge fittings.

Fuel bulkheads will not be required in the horizontal stabilizer, but one bulkhead is required at the centerline splice between right- and left-hand stabilizers.

The vertical stabilizer skins are automatically tape laid, densified, and staged, Figure 30. The aluminum honeycomb is EDM machined, Figure 31. The densified and staged skins, adhesive, and EDM-machined aluminum honeycomb core are assembled on the PLM. The vertical stabilizer skins are cocured and bonded to the honeycomb core at 350°F and 40-psig autoclave pressure for 2 hours, Figure 32, after pultruded T-section spar caps have been included in the assembly layup. Rib and center spar sandwich panels are made in a similar manner (Figures 30, 31, and 32). The derby-hat-stiffened front and rear shear webs are fabricated in a manner similar to the wing and horizontal tail box webs (Figure 23).

Assembly and bonding of the vertical stabilizer substructure and the individual dual inflatable mandrels for each compartment are illustrated in Figure 33, along with the aluminum leading-edge components. Figure 33 also illustrates the rudder assembly procedure which is typical also for elevators, spoilers, and ailerons.

The forward and aft upper and lower graphite/epoxy rudder skins are automatically tape laid, as shown for the stabilizer skins (Figure 30). The aft rudder wedge-shaped substructure components are hand laid and cured as shown in Figure 34. The forward rudders will be fabricated in a manner similar to the aft rudders, except a rear spar with openings in it to allow push-rod controls will be incorporated.

Figure 35 illustrates the pultrusion, densifying, and staging molds for the fabrication of vertical stabilizer trailing edge access panels and tapered T-spar caps.

The access panel doors are pultruded to their proper length, which includes a raised picture-frame edge buildup. Pultrusion die pressure is maintained by passing filler plugs through the die on a continuous conveyor belt slip sheet that circulates through the die, Figure 35. The filler plugs are indexed to fill the openings, or depressions, between the picture frames.

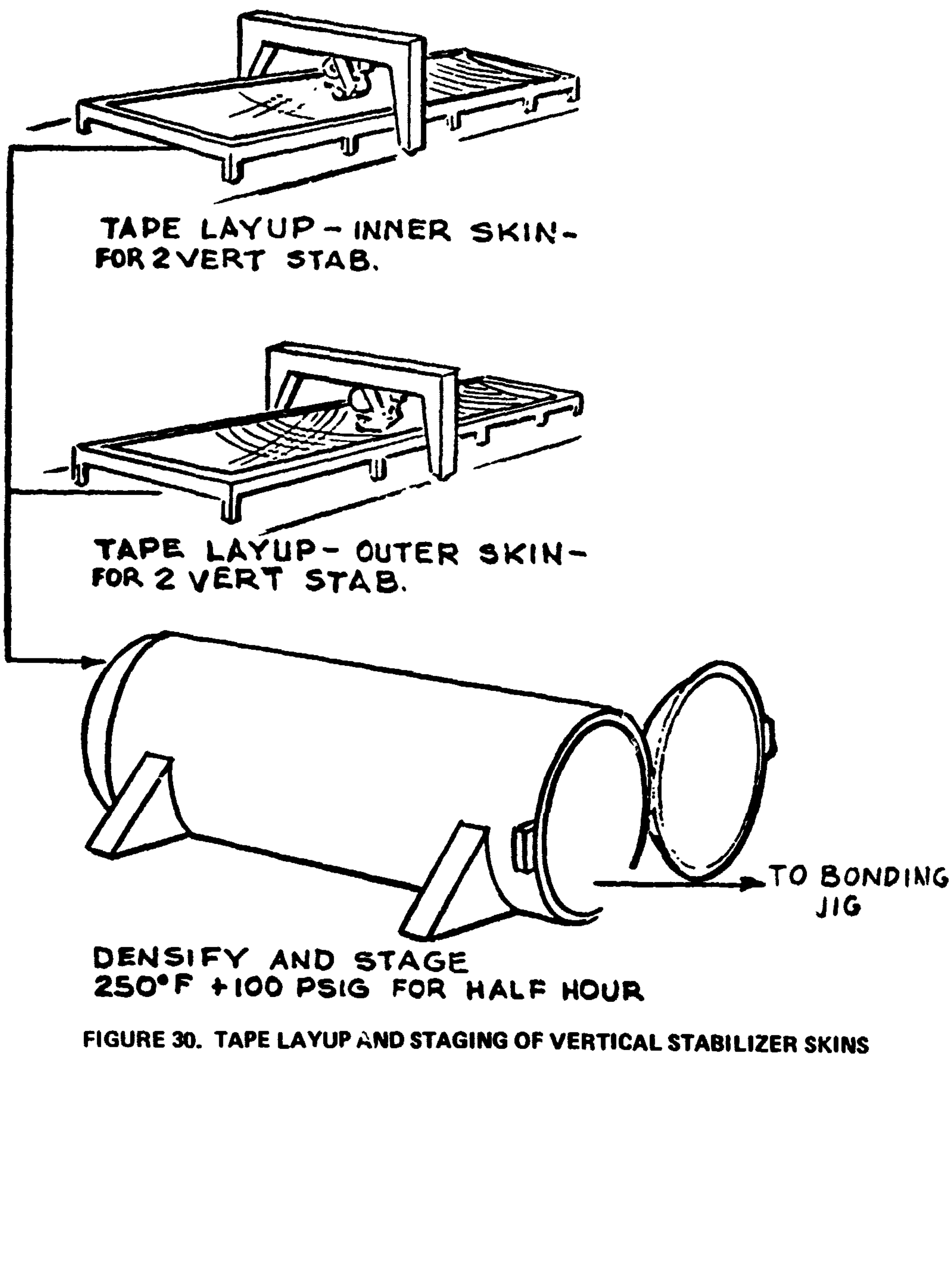


FIGURE 30. TAPE LAYUP AND STAGING OF VERTICAL STABILIZER SKINS

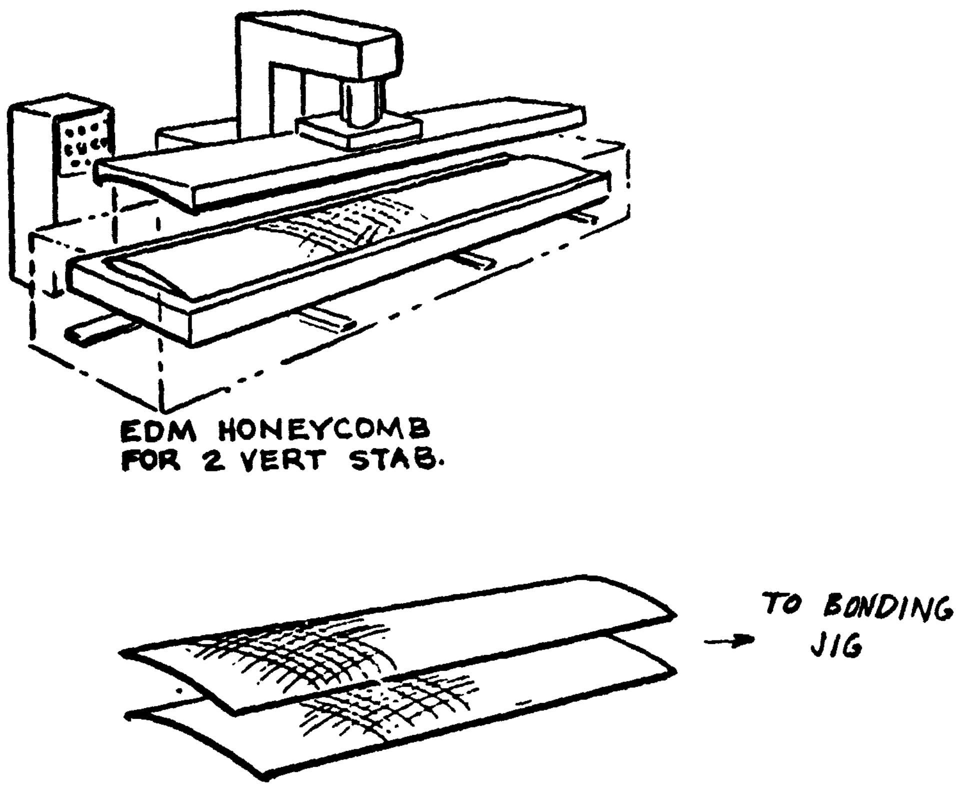


FIGURE 31. EDM MACHINING OF ALUMINUM HONEYCOMB CORE

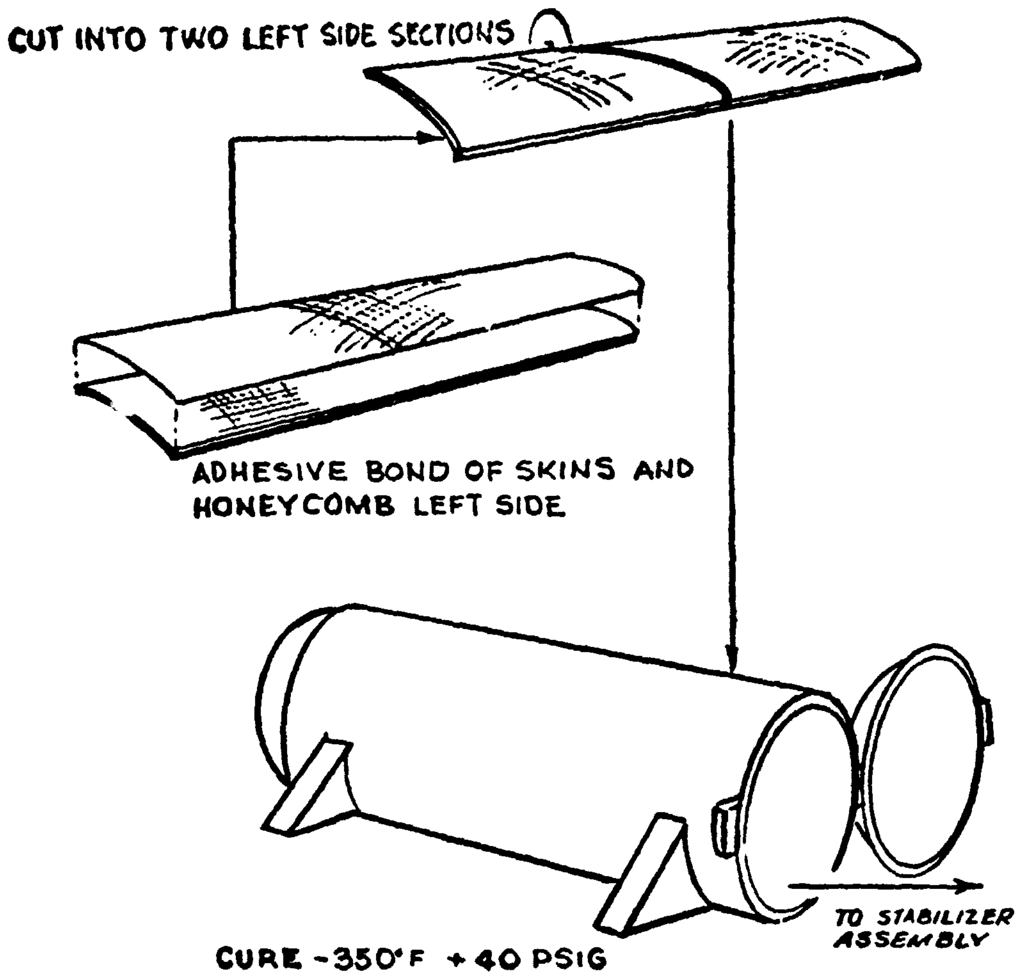


FIGURE 32. COCURING AND BONDING VERTICAL STABILIZER SKINS TO THE ALUMINUM HONEYCOMB CORE

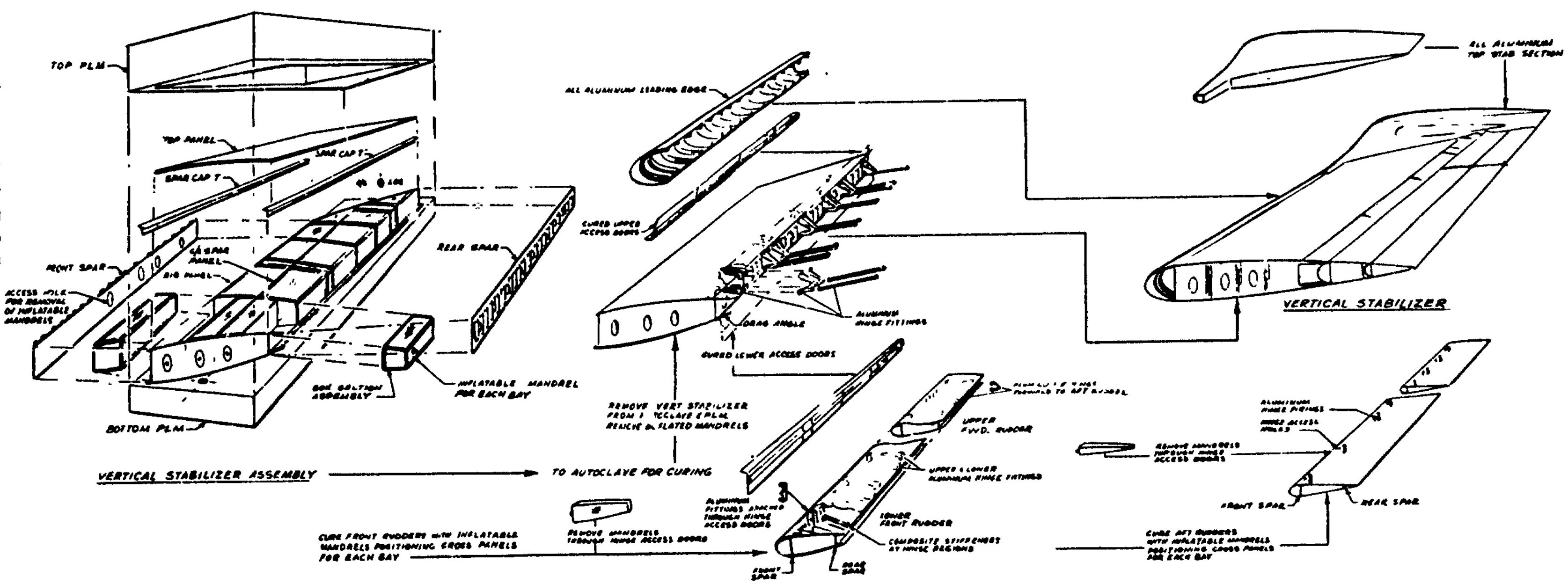


FIGURE 33. VERTICAL STABILIZER MANUFACTURING ASSEMBLY

BEST AVAILABLE COPY

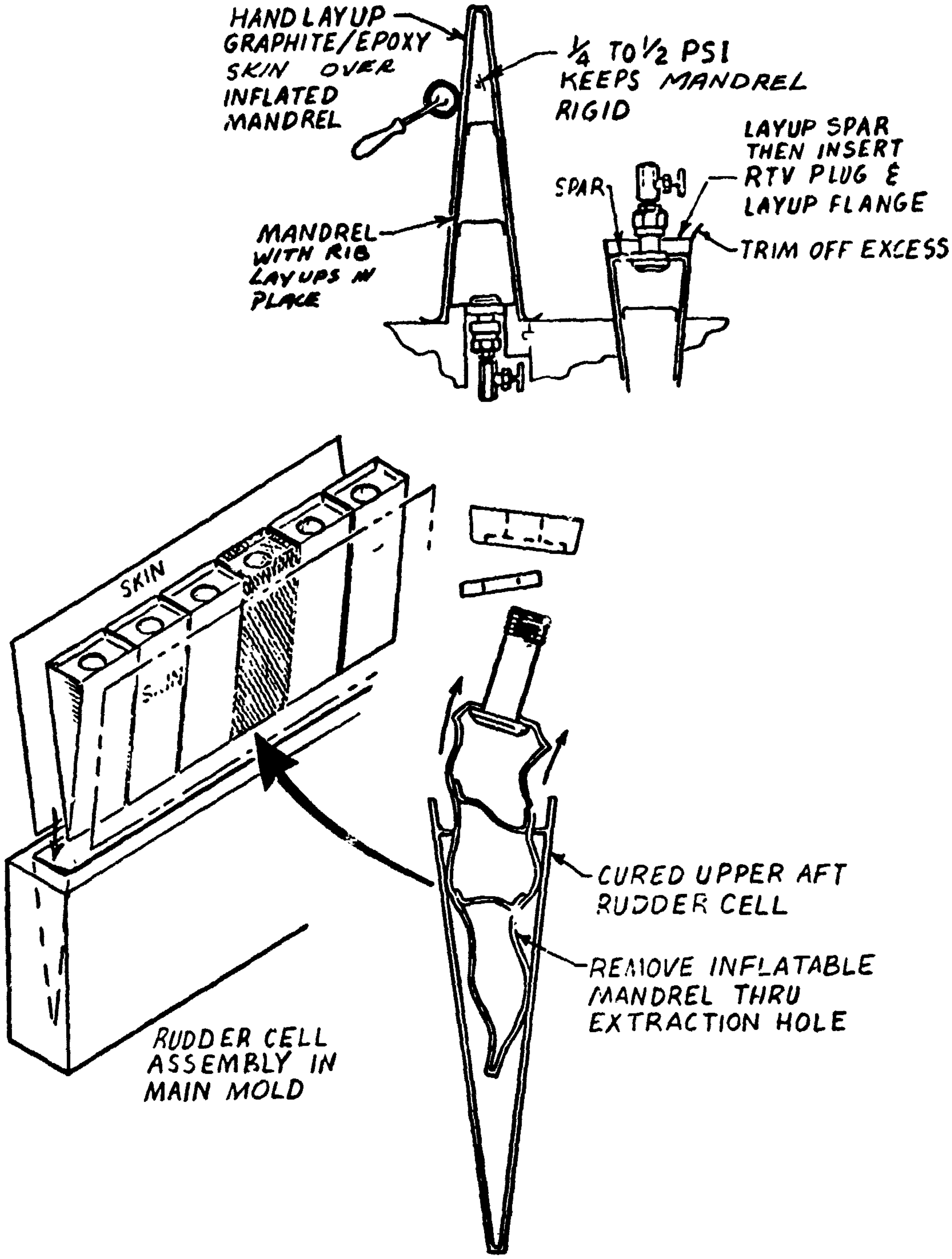


FIGURE 34. AFT RUDDER FABRICATION AND TOOL REMOVAL

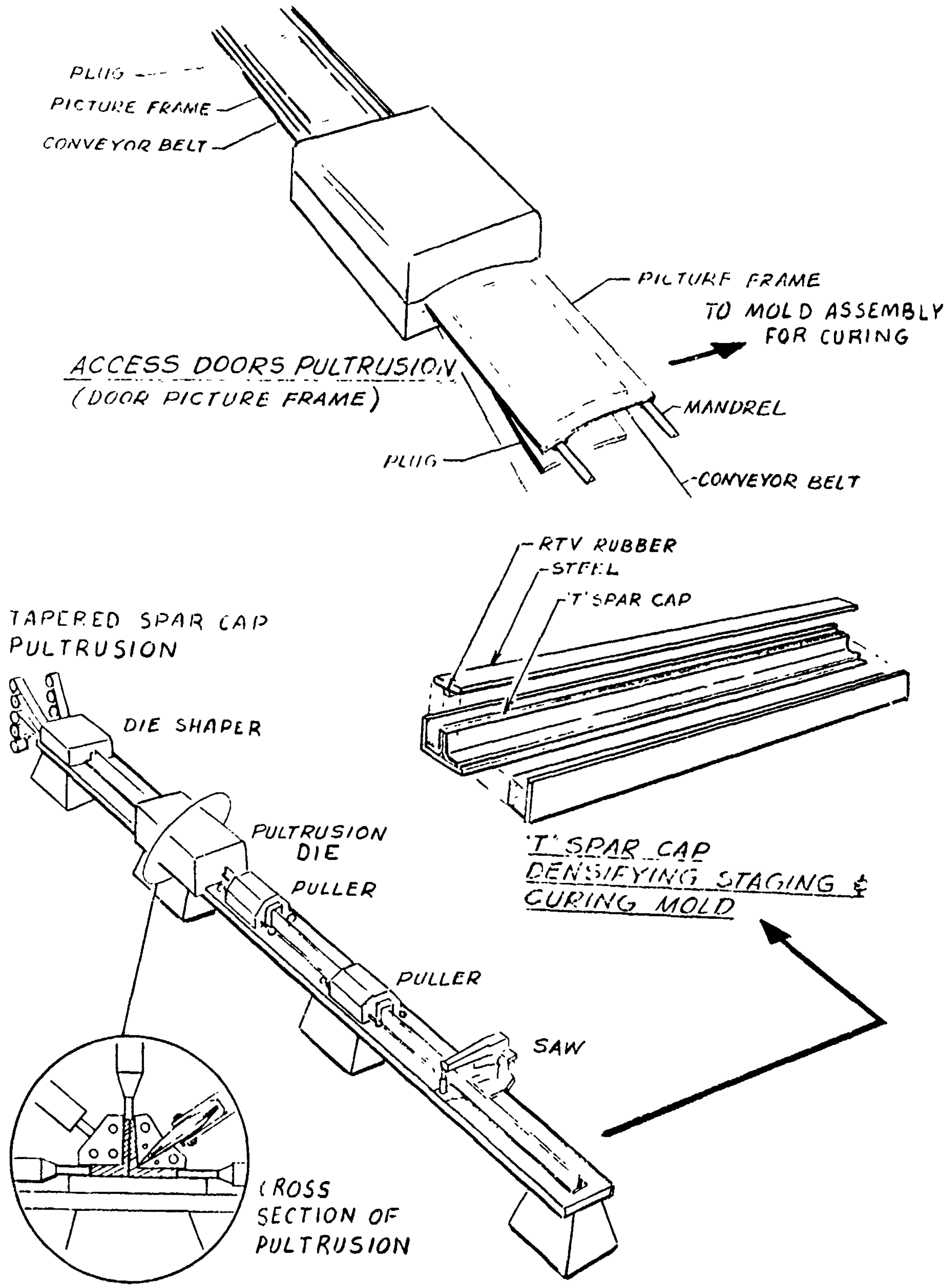


FIGURE 35. PULTRUSION OF TRAILING EDGE ACCESS DOORS AND TAPERED SPAR CAPS

The T-spar caps are tapered lengthwise in cross section. This is accomplished by having style 1050 prepreg graphite slit to the exact taper, widths, and lengths to make the end item. The cross section view of the pultrusion die, Figure 35, shows how the variable die is adjusted to maintain shape and pressure. The pultruded components are placed in densifying and staging dies and cured for 2 hours at 350°F and 100-psi pressure.

Upon completion, the vertical stabilizer and rudders are forwarded to the next assembly station.

4.3.3 Composite Fuselage Manufacturing and Assembly Outline.

The main fuselage barrel, being cylindrical, has a constant cross section. The winding technique utilized for fabrication of the shell is illustrated in Figure 36. A machine with multiple band feed capability and computer control is required. Numerical control does not offer a self-correcting capability. The machine utilizes composite material bands (slit tape or broadgoods) as wide as the isogrid ribs, in this case, 0.10 inch.

Isogrid Shell Wrapping Process - The constant section fuselage wrapping process is believed to be completely feasible. The scheme includes turnaround of each band at each end of the cylinder to produce the V-joints pictured in View J-J of Figure C-2. Figure 37 illustrates, in exaggerated perspective, how multiple head wrapping circuits return on themselves in the case of the ±30-degree grid ribs in an arbitrary cylinder, so that a complete layer of tape is laid in all mandrel grooves in each wrapping cycle. In the case shown, 4 wrapping heads make 12 traverses of the mandrel per cycle to produce a complete layer with 48 circumferential divisions. Other integral relationships exist. If a full circumference of wrapping heads (48) is used, only one traverse of the mandrel down and back completes a layer. In the fuselage constant section design case, a full circumference of 144 wrapping heads is used in 127 subdivisions between Stations 439 and 947. This eliminates the necessity of having the number of longitudinal subdivisions divisible by the number of wrapping heads.

The aft fuselage grid, being tapered, could be wrapped with the same number of circumferential subdivisions per station to the tailcone aft joining station. Figure 37 can be used to illustrate this case if it is viewed as the projection of a frustum of a right cone (approximating an aft fuselage shape). This produces, however, too great a grid weight penalty so the idea of dropping off grid circumferential subdivisions was introduced. This would be accomplished by reversing direction of some, but not all, of the 144 wrapping heads in mid-cycle, much as all the heads are reversed at cylinder ends. This is illustrated schematically in Figure 38.

Figure 38 is an arbitrary cone frustum (it could be an oblique cone to better approximate the aft fuselage shape) with 48 circumferential subdivisions at Stations 0, 1, and 2. Six wrapping heads are first assumed in the example. Heads 1, 3, and 5 reverse at Station 2 and return to Station 0 as Heads 2, 4, and 6 continue to wrap down to Station 4 (there are 24 subdivisions of Stations 3 and 4), and then reverse. Meanwhile, Heads 1, 3, and 5 have indexed at Station 0, moving with the mandrel rotation without laying tape until the set 2, 4, and 6 has also returned, completing one traverse. Prior to wrapping, inserts have been installed on the mandrel, aligned along cone elements

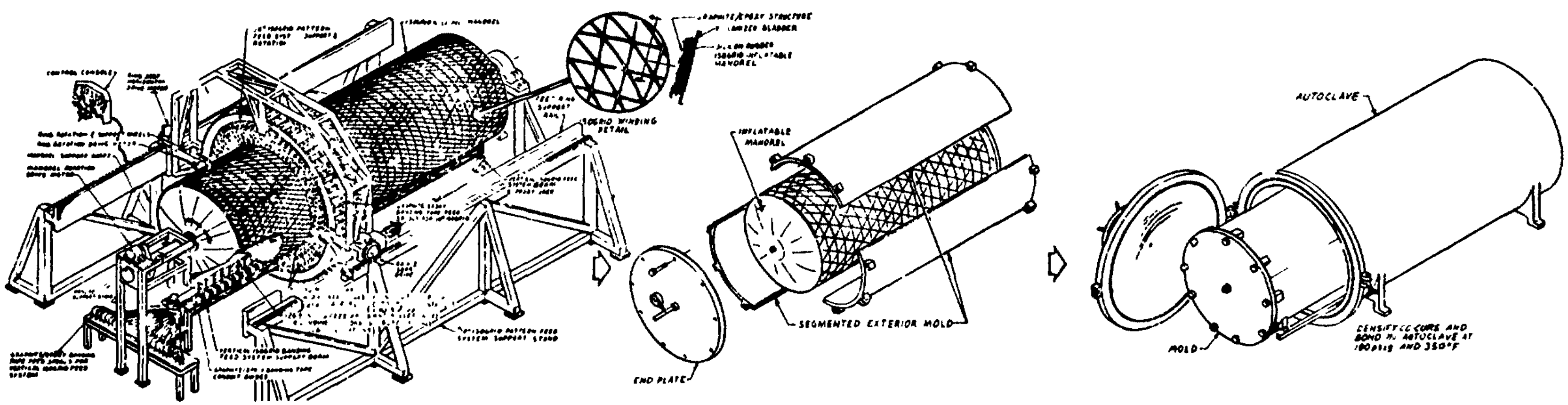


FIGURE 36. ISOGRID FABRICATION CONCEPT IV

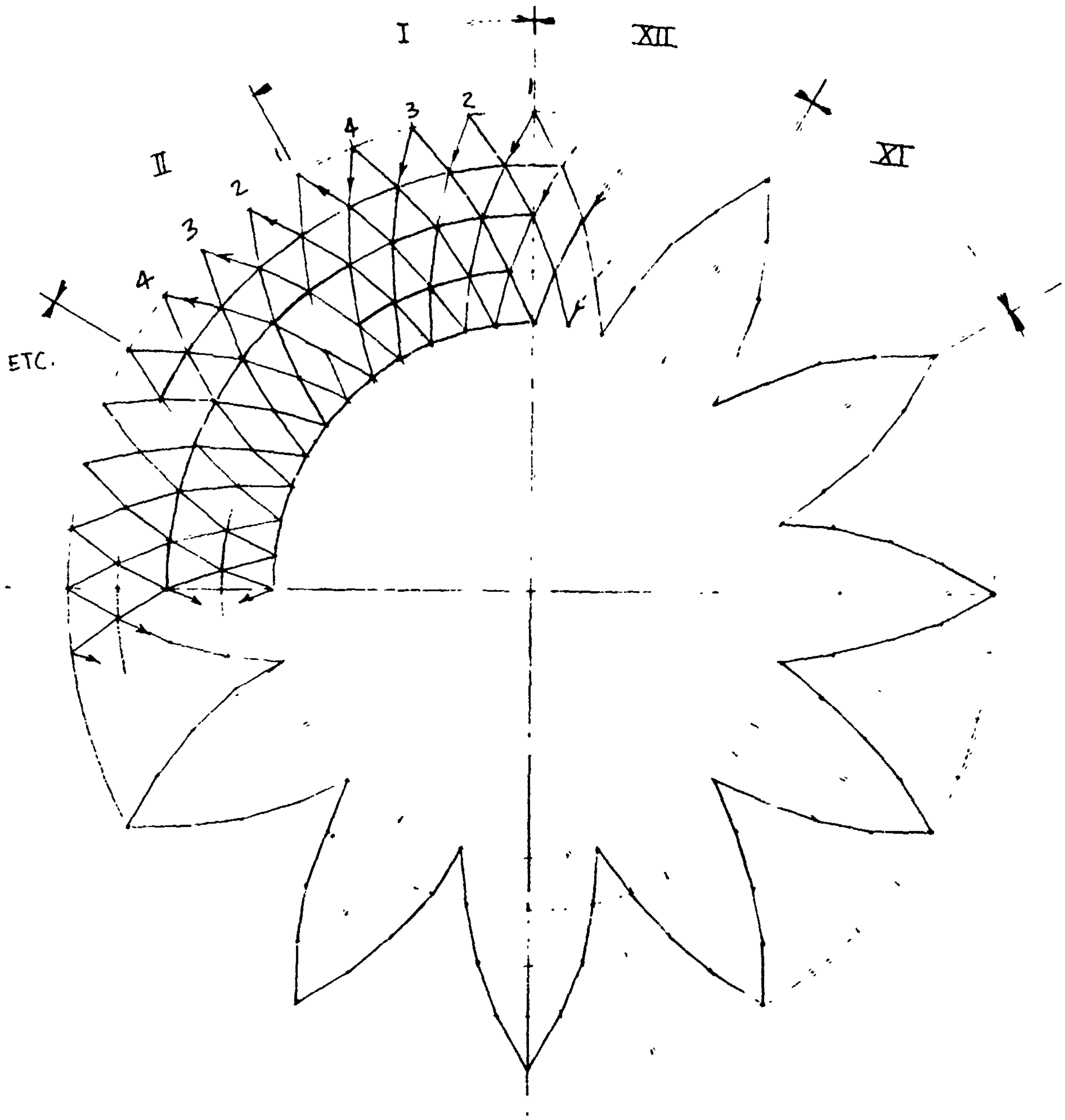


FIGURE 37. SCHEMATIC PERSPECTIVE OF ARBITRARY WRAPPING PATTERN FOR A SHORT CYLINDER

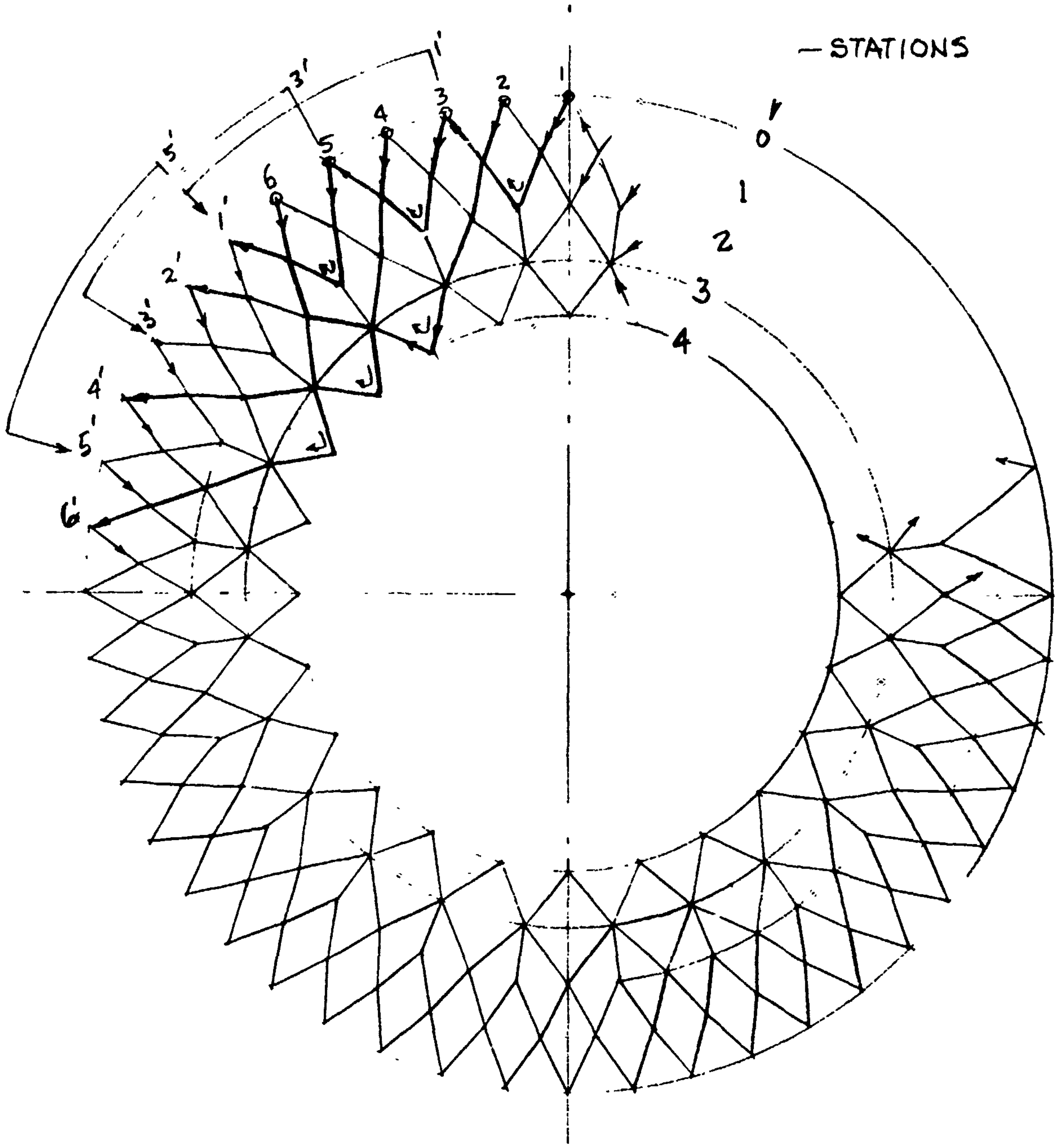


FIGURE 38. SCHEMATIC OF CONE FRUSTUM WRAPPED WITH DECREASED NUMBER OF CIRCUMFERENTIAL SUBDIVISIONS

between Stations 2 and 3 to connect triangle vertices where triangles are dropped out of the pattern. Three sets of two heads, as in the example (Figure 38), will wrap a full coverage longitudinal layer in 8 traverses. At this point, the 90-degree layers would be wrapped at the station planes and the cycle would repeat for additional layers.

Wrapping the example cone with 6 or 9 sets of heads results in 2 and 3 complete layers, respectively, being deposited in 8 traverses, with the obvious extension to more wrapping heads and more layers. Unfortunately, with multiple traverses laying multiple layers, there is no opportunity to intersperse 90-degree wraps to produce the characteristic composite isogrid joint. Conversely, using less than 48 wrapping heads and fewer than 8 mandrel traverses results in incomplete coverage of the mandrel.

For the specific 48-subdivision case shown, a single traverse (Stations 0 to 4 and back) with 24 sets of heads (2 heads each set) produces one full layer. A full circumference of wrapping heads used for a single traverse per layer is thus substantiable in the general case in order to minimize both wrapping time and the complexity of wrapping geometry.

The mechanization of the scheme is believed feasible but requires additional development work for verification. Cost and weight estimates are based on the use of the general fabrication concept.

Since the forward and aft fuselage manufacturing techniques are basically the same, except for the extra generated motions required for winding the tapered section, the aft section fabrication is outlined in greater detail here.

V- and attach-angle inserts for the cylinder end joints are automatically installed into forward and aft fuselage inflatable mandrels, as shown in Figure 39. The end joint inserts that go between the splice ring and the isogrid triangles, Figure 39, are installed after the substructure winding operation, prior to winding the fuselage splice ring and outer skin. The triangular bolting blocks that fit on the inside of the isogrid at the extreme forward and aft ends of each fuselage section are installed after the fuselage has been cured and the inflatable mandrel removed, Figure 39.

The major differences between winding the aft and forward sections are shown in Figure 40, which illustrates a circumferential 3-piece split ring. This 3-in-1 ring allows shifting of the 144 individual banding feed arms and the motion-picture-type spools containing preimpregnated graphite/epoxy material. This allows the station-to-station winding and reversing of any number of the 144 feed arms in any one of the 3-ring segments as required by wrapping geometry.

Table 8 illustrates the fuselage stations, approximate diameters, number of circumferential triangles, and variation in length of the 90-degree legs of the triangles. The equilateral triangles at Station 947 become isosceles triangles as the fuselage begins to taper.

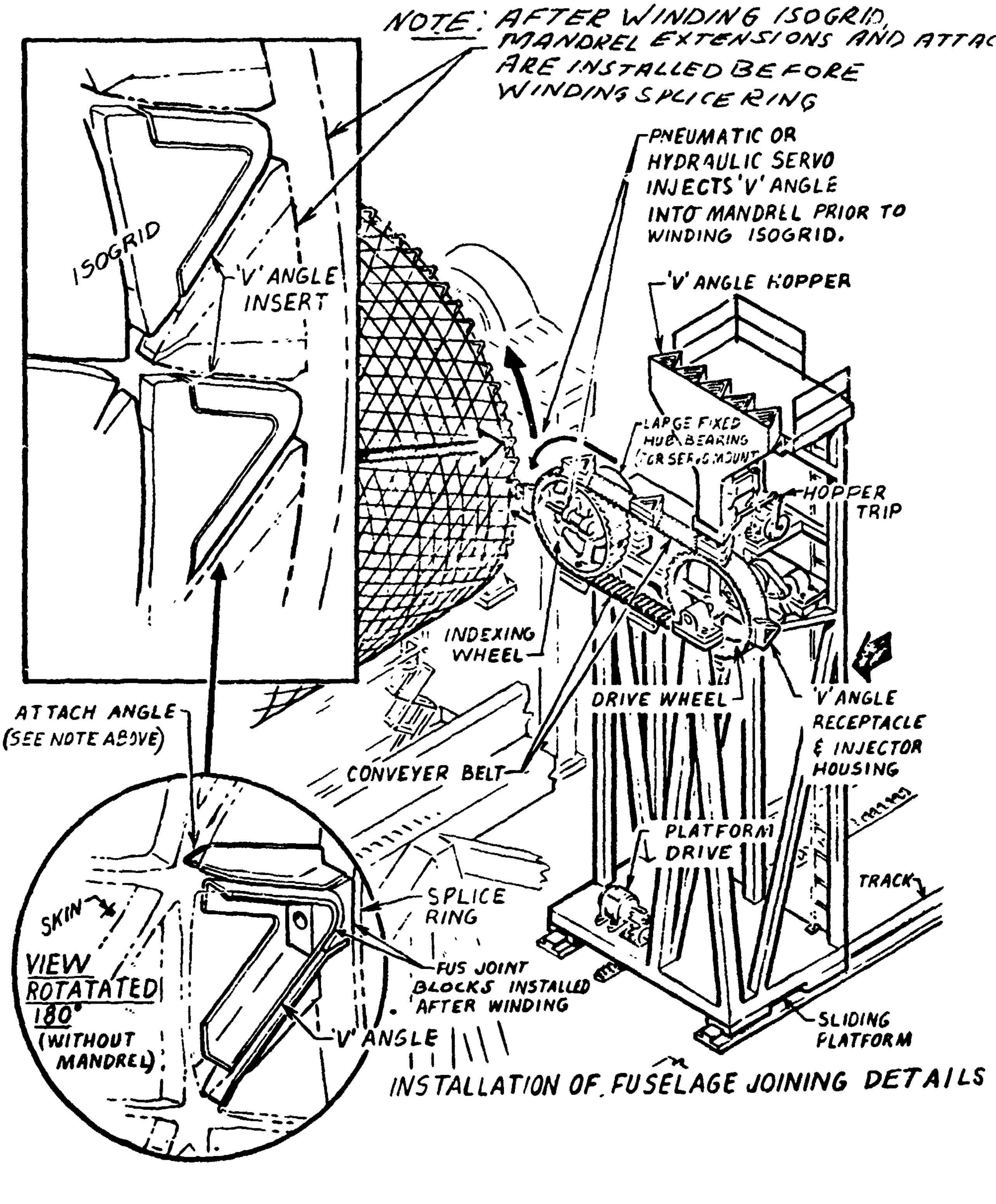


FIGURE 39. INSTALLATION OF FUSELAGE JOINING DETAILS

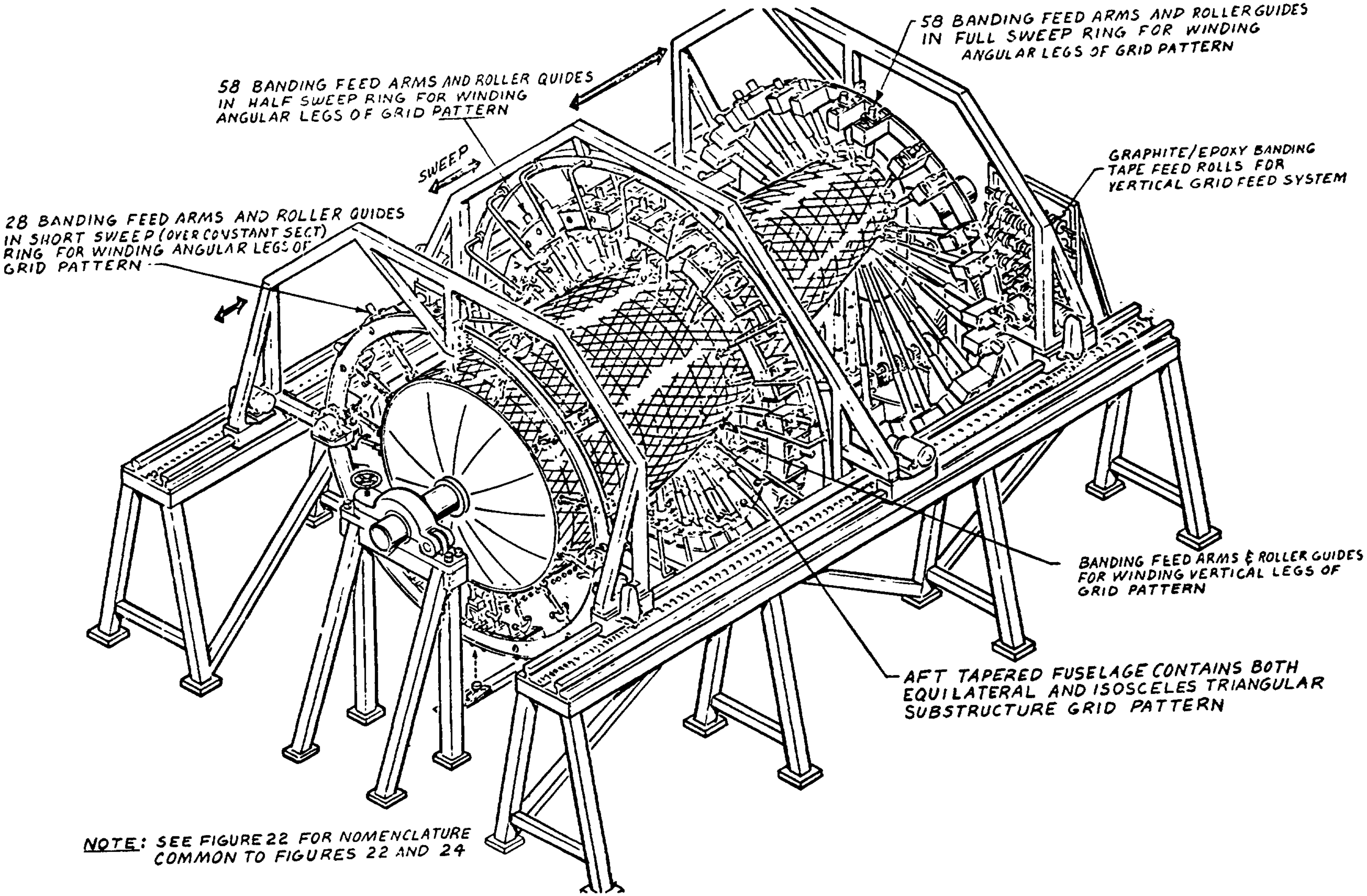


FIGURE 40. WINDING AFT FUSELAGE

TABLE 8

TAPERED AFT FUSELAGE TOOL DESIGN DATA

FUSELAGE STATION (INCHES)	APPROXIMATE DIAMETER (INCHES)	NUMBER OF TRIANGLES	90-DEGREE LEG LENGTH (INCHES)
947	216.00	144	4.62
1127*	186.00	144	4.05
1247*	146.00	144	3.19
1251	144.75	116	3.92
1343*	116.00	116	3.13
1347	113.92	58	6.17
1439*	66.00	58	3.59

*TURNAROUND STATIONS

Although Station 1127 is a designated turnaround station, the number of triangles at the next turnaround station (1247) is still 144 because triangles were not dropped at Station 1127. All 144 heads reverse at Station 1127 for the layers laid first on the mandrel. This produces full grid depth in the region of highest shell stresses, Stations 947 to 1127. On succeeding layers, the full length of the aft fuselage is wrapped with turnaround of appropriate numbers of heads at Stations 1247 and 1343, thus producing less grid depth in those aft areas with a reduced structural stability requirement.

The reversing stations (where different numbers of banding arms and a segment of the circumferential ring change directions) can be varied, but must remain an integral function of the station-to-station distances of the ± 30-degree intersections (4.0 inches). This system flexibility allows the grid depth to be varied for different structural requirements as noted above.

Since the number of triangles in the circumferential direction is reduced as the tapered fuselage decreases in diameter, a void is created between the turnaround point and the next inline triangle in the longitudinal direction. Before wrapping, metal or composite inserts are installed on the inflatable mandrel to complete the unfinished triangular legs of the grid just aft of the reversing stations. Wedge-shaped inserts also help reduce stress concentration points at the change in depth of the isogrid/isosceles triangles (Station 1127).

The ± 30-degree isogrid/isosceles triangular legs will be wound at a surface velocity of 20 feet per minute. Each segment of the circumferential rings will be indexed with, but independent of, the other two rings, as reverses and changes of direction occur.

Upon completion of a longitudinal pass in each direction along the inflatable mandrel, each ring segment will lock in place at the forward end and rotate with the mandrel. This indexing and locking dwell is essential to allow the remaining two segments of the circumferential ring to complete their longer traverses. When all segments of the ring have completed their traverses, the 90-degree layer of graphite/epoxy tapes is applied and then the cycle repeats.

Midway through the winding operation, the glass stress concentration relief inserts are installed in each end joint. Winding is then resumed.

Banding Control - During the winding of the tapes there is a continuous spreading and contraction of the three tapes on each of the 144 banding heads. This is illustrated conceptually in Figures 41 and 42. Figure 43 demonstrates the banding system setup for winding the 90-degree legs of the grid. Band control is also maintained for the turnaround stations and joints.

Skin Wrapping - After completion of the tapered fuselage substructure winding, the 90-degree and the segmented-ring feed arms (Figure 40) are reloaded with full reels of graphite/epoxy tapes, 0.10 inch wide. Each feed arm contains three double reels 24 inches in diameter with continuous tape approximately 20,000 feet long.

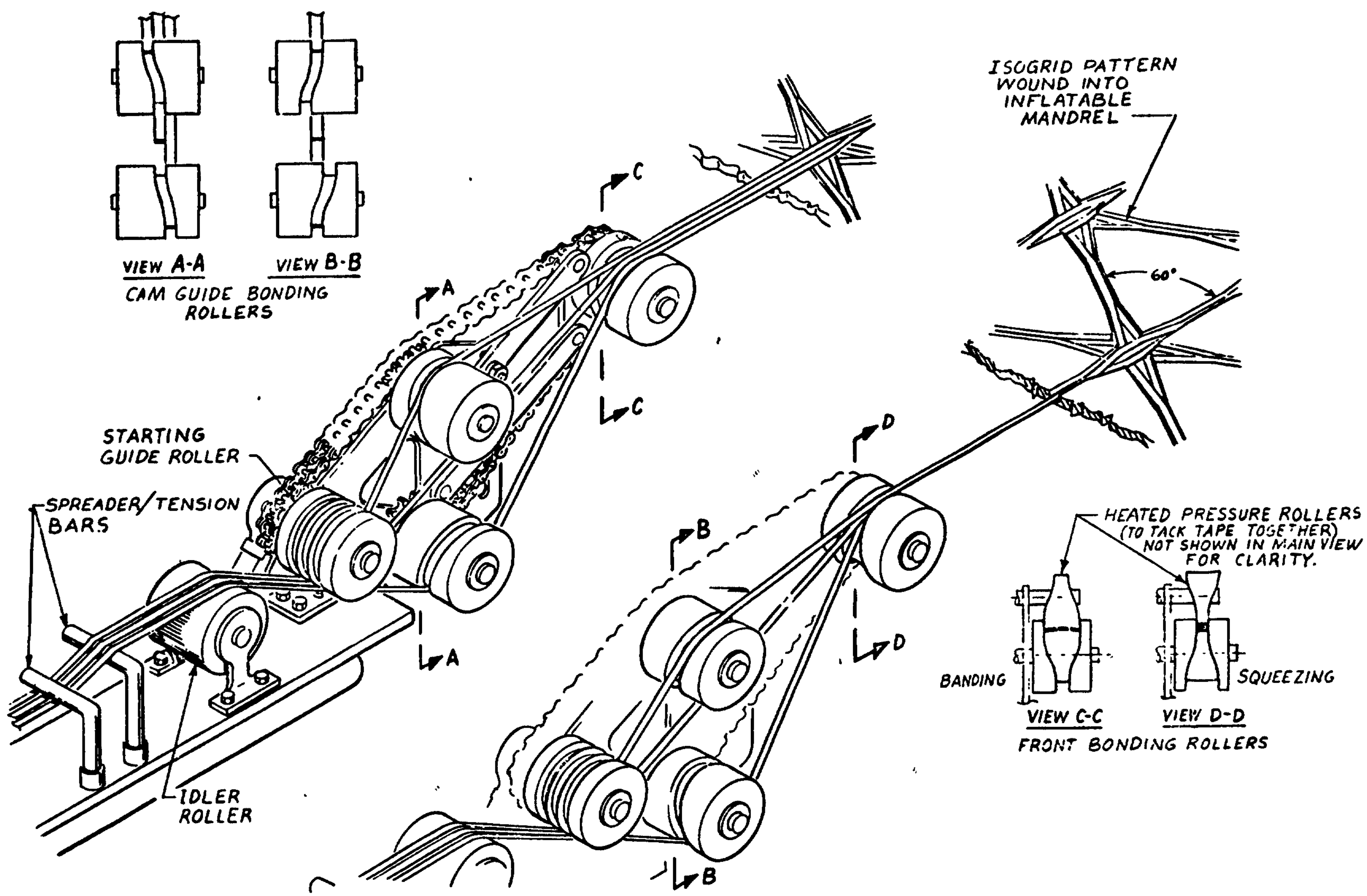


FIGURE 41. ISOGRID WINDING/BANDING SYSTEM CONCEPT III

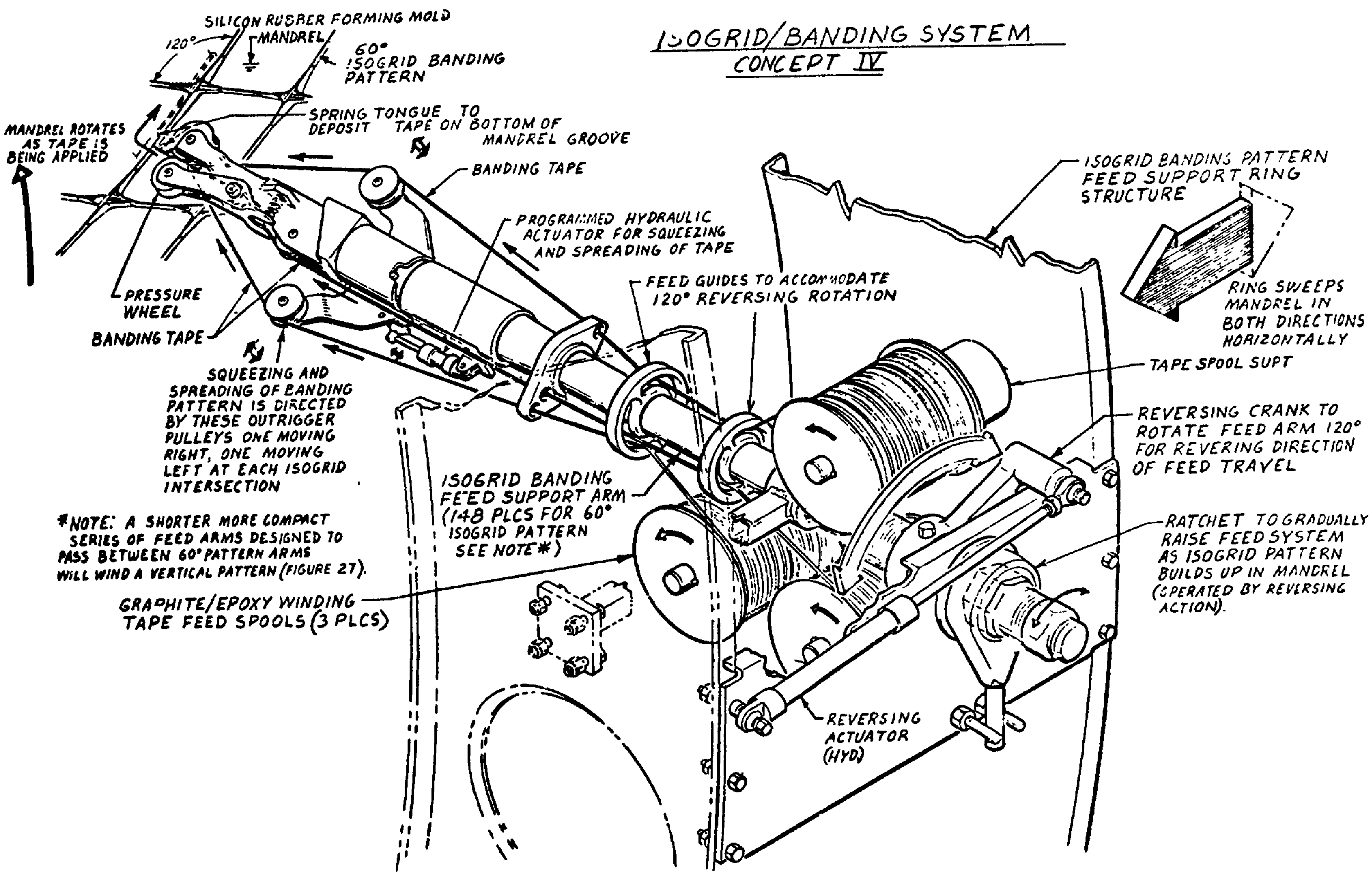


FIGURE 42. ISOGRID/BANDING SYSTEM CONCEPT IV

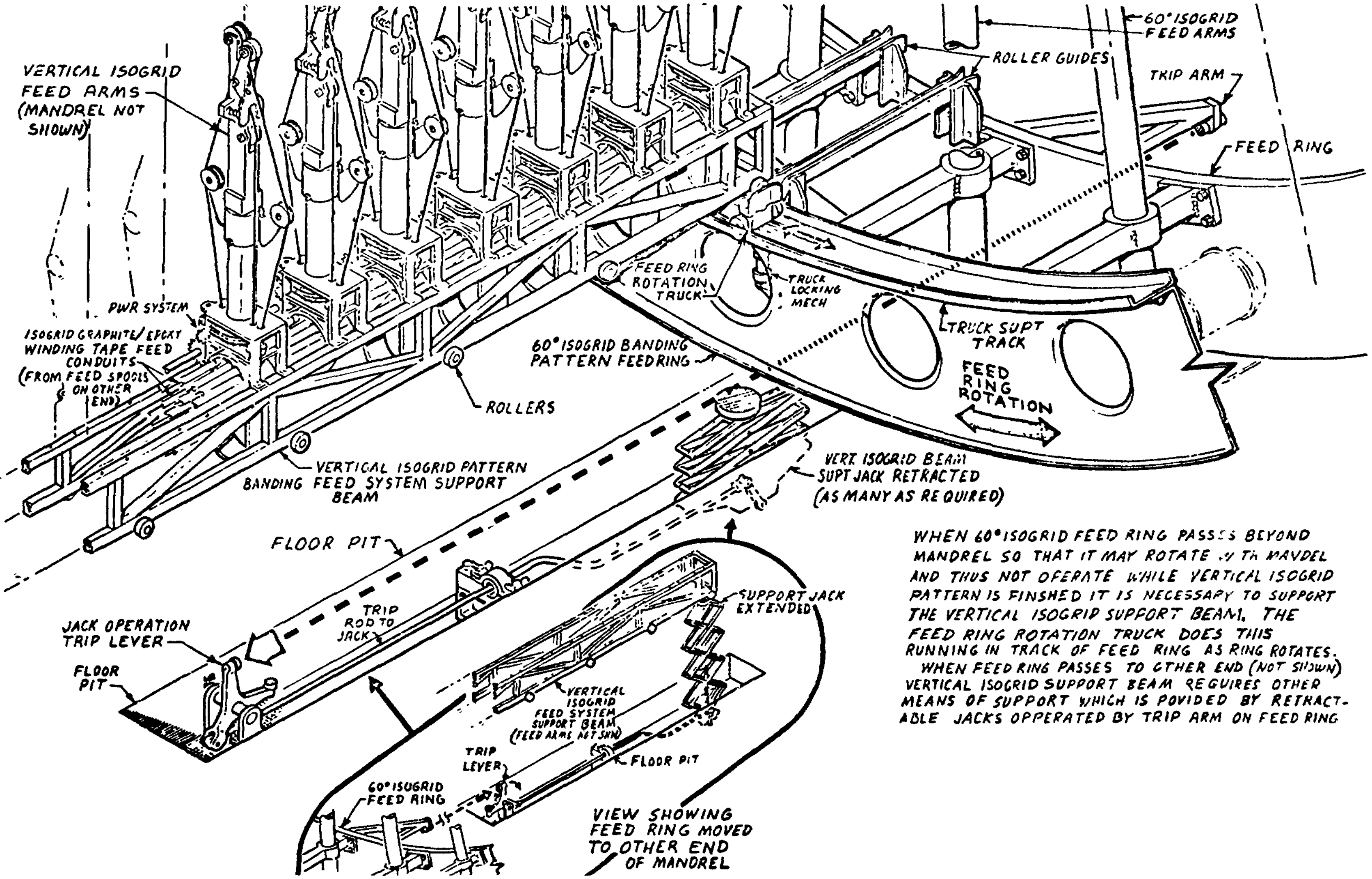


FIGURE 43. ISOGRID/BANDING SYSTEM CONCEPT IV

The feed arms can be set up to include winding-in fiberglass crack-stop strips between the graphite/epoxy bands, in accordance with Design Engineering requirements.

Based on using the 144 ring stations and maintaining a constant 0.30-inch-wide tape for each bander, the fuselage outer skin could be wound in a very short period of time as a result of the wide dispersal of material. The main cylindrical section of the fuselage can be skinned in this way. The skin on the tapered aft fuselage would have to be wrapped with over-lapping bands and constantly changing wrap angles, which is unacceptable from weight and wrapping control complexity standpoints. Therefore, if a method for dropping out bands sequentially to taper the skin sheet width is not feasible, gore sections cut from flat machine layups will be transferred to the fuselage layup on peel-off support sheets. The 90-degree layup direction can be laid up directly on the aft fuselage using the grid wrapping machine.

After winding the initial layers of the fuselage skin, reinforcement inserts for windows, doors, fittings, and floor attach points are indexed and fastened to the structure. The winding operation of the fuselage skin is continued to completion which includes the circumferential splice rings for joining to adjacent fuselage sections as shown in Figure 44.

Curing the Fuselage Barrels - The segmented exterior mold is installed around the "B" staged fuselage composite structure (Figure 36). The entire structure and mold are placed in an autoclave. The autoclave and inflatable mandrel pressures are increased to 25 psig. The exterior mold is sealed and vacuum is held while venting the inflatable mandrel to the autoclave pressure. The autoclave temperature is increased at a rate of 2 to 9°F per minute, while increasing the pressure to 100 psig. Pressure is maintained for 4 hours at 350°F. Upon completion of the cure cycle at temperature, autoclave pressure is decreased to 25 psig while the temperature is lowered to 300°F. The inflatable mandrel is reinflated to 25 psig. The exterior mold is automatically released to allow for thermal contraction of the aluminum shell as it cools to room temperature. The autoclave and inflatable mandrel pressures are reduced to 5 psig. When the temperature reduces to below 150°F, the autoclave pressure is reduced to atmospheric, but the inflatable mandrel pressure is maintained at 5 psig.

The fuselage and the segmented exterior mold are removed from the autoclave. After removing the deflated mandrel, the structure is prepared for installation of bulkheads, fittings, large frames, etc.

Aft Fuselage Space Frame - The space frame tubing, Figure 45, is 1.5-inch-OD graphite/epoxy with 0.065-inch wall thickness. The tubing and the connecting frame cap T-sections are pultruded and cured sufficiently, while in the pultrusion die, to allow an oven postcure without support of the parts.

The graphite/epoxy space frame joint fittings, Figure 45, are molded to net size and cured. The space-frame panels with elliptical lightening holes are press molded and cured to net size. The space-frame components are assembled in modular segments prior to installation into the aft fuselage, Figure 46. In like manner the door area torque box components are pultruded and molded prior to assembly into the aft fuselage, Figure 47.

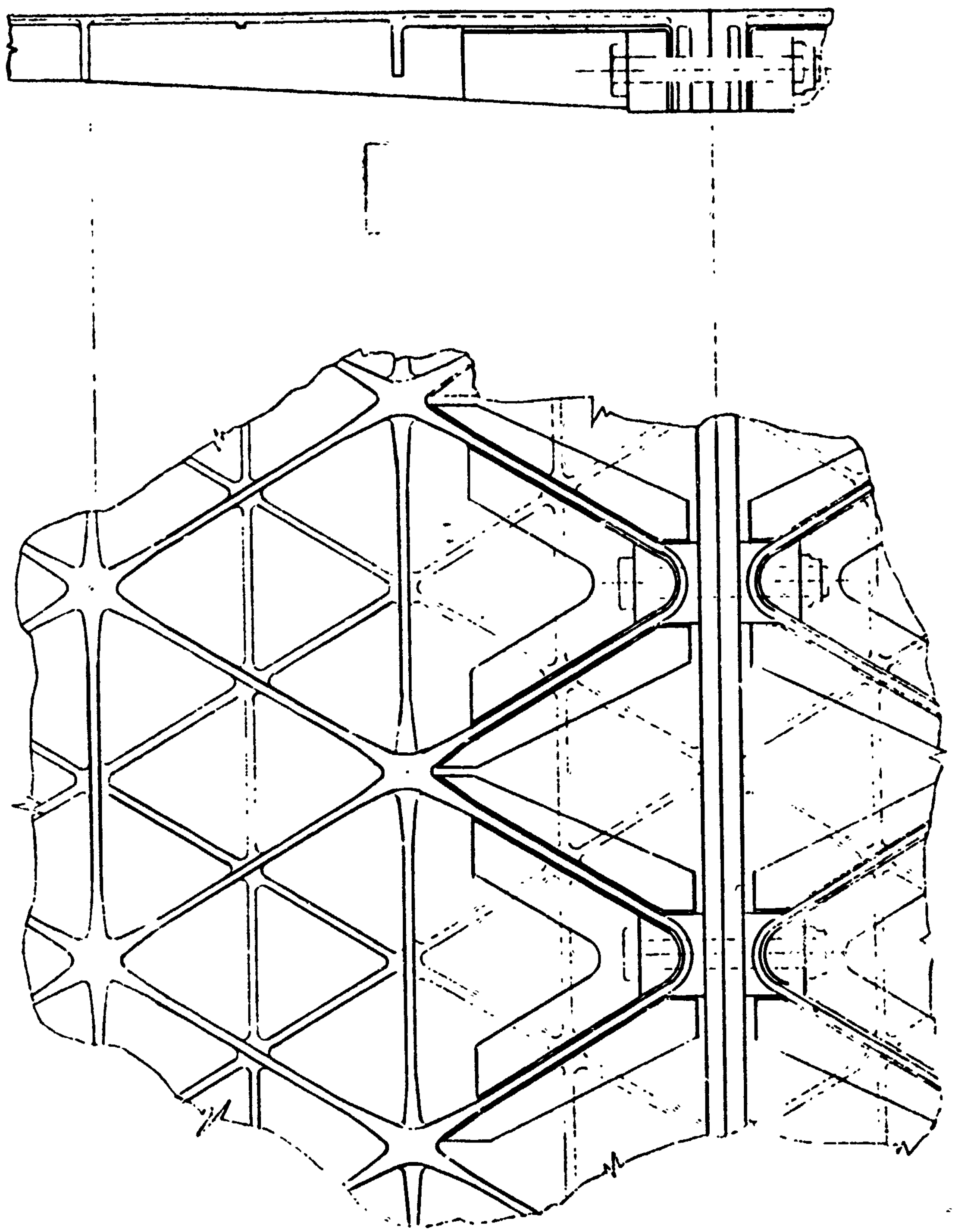


FIGURE 44. FORWARD AND AFT FUSELAGE CIRCUMFERENTIAL SPLICE RING JOINT

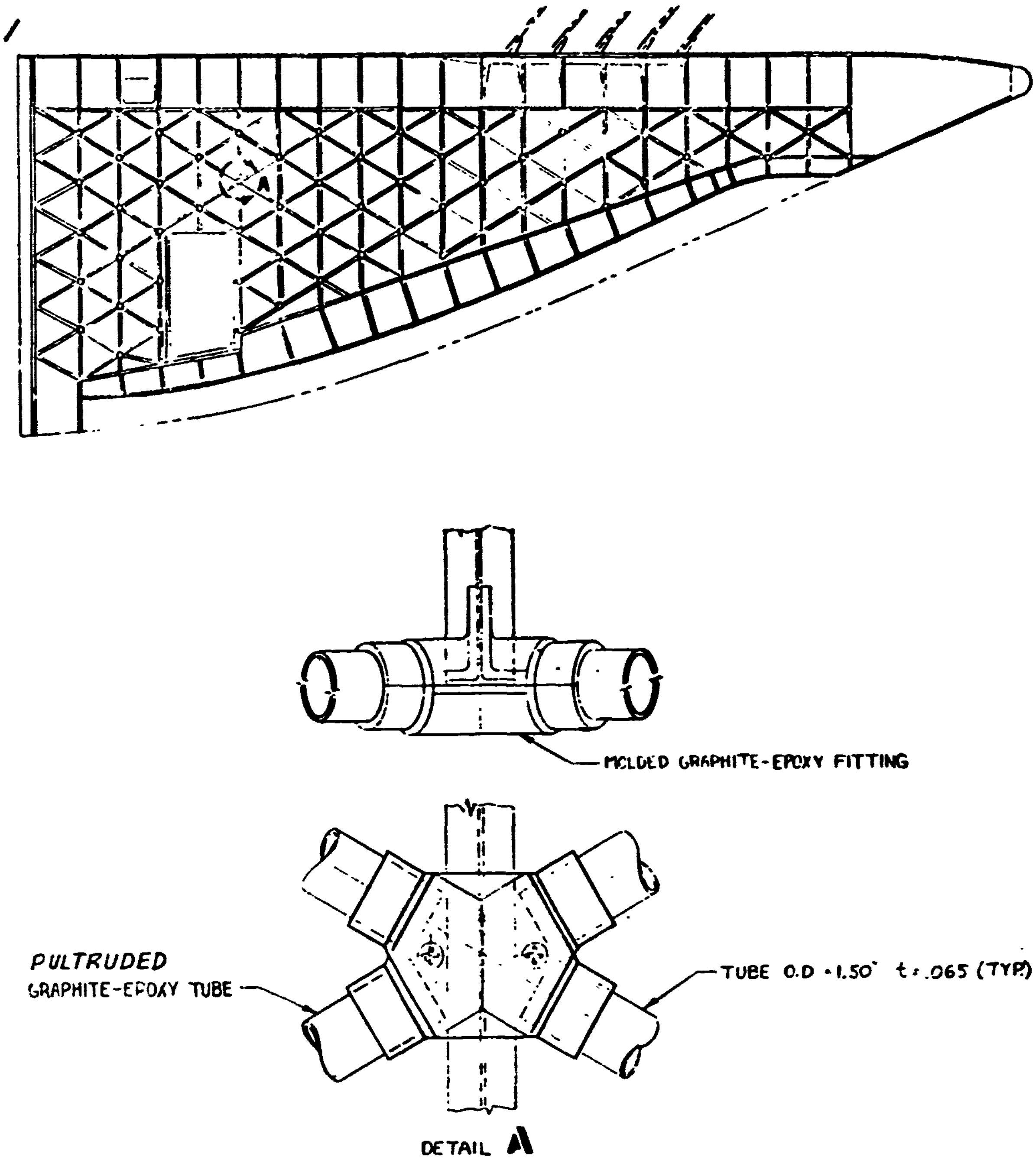


FIGURE 45. TYPICAL SPACE FRAME JOINT

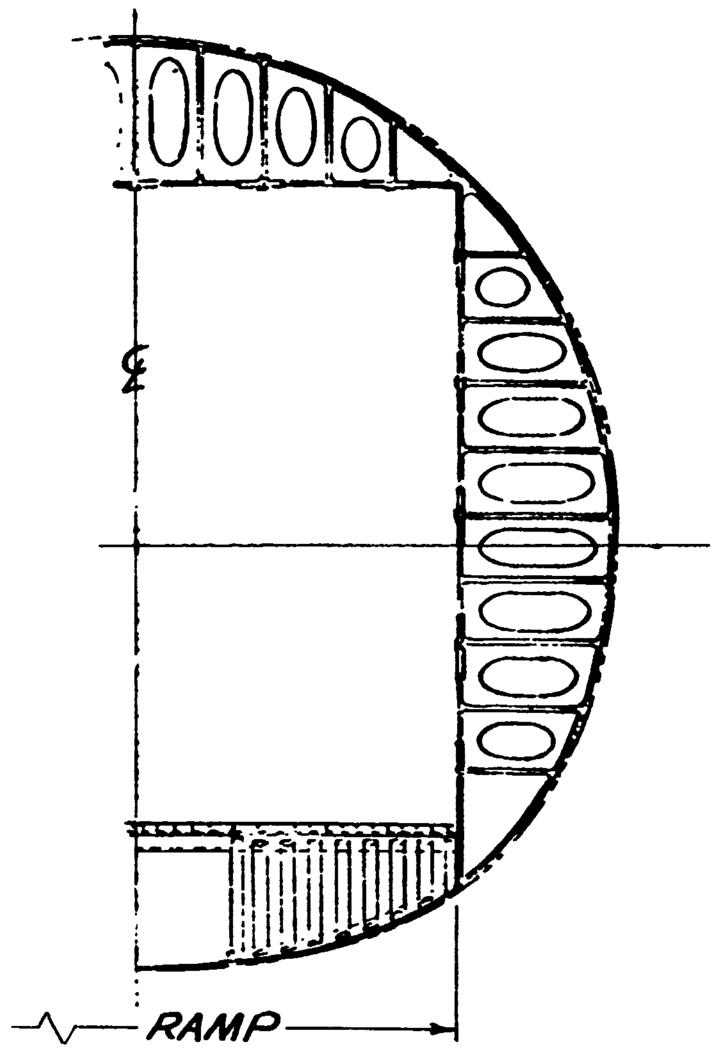


FIGURE 46. AFT SECTION FRAME AND RAMP CROSS SECTION

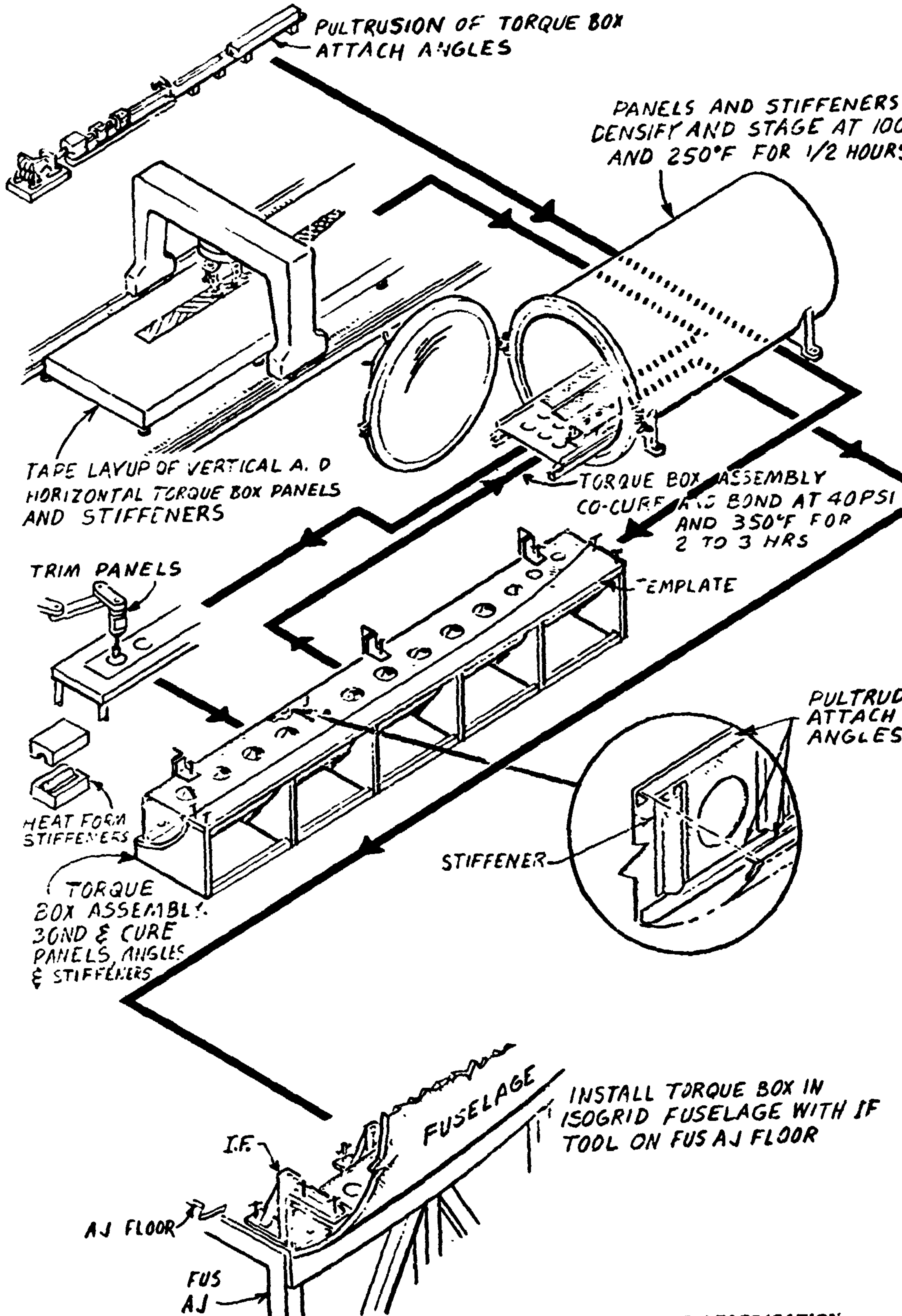


FIGURE 47. AFT CARGO LOADING AREA TORQUE BOX FABRICATION

Fuselage Subassemblies - The section of the aft fuselage shell that was cut out to provide the opening for the cargo loading ramp and loading door becomes the stiffened skin from which the ramp and door are constructed. Edge torque boxes, ramp floor supports, ramp flooring, door stiffening, latches, and hinges are added to complete these major subassemblies.

The floor support and keel beam subassembly is installed into the forward fuselage shell as shown in Figure 48. Drill jigs and routing templates are used to install all hinge fittings, door frames, and latches. Doors and ramps are installed and adjusted prior to release of the fuselage sections to final assembly.

Fuselage Nose Manufacturing and Assembly Outline - Two fuselage nose section outer skins are simultaneously tape wound, densified, and staged in an autoclave at 100 psig and 250°F for one-half hour. See Figure 49 for fuselage nose fabrication sequences. Figure 49 illustrates the separation of the two outer skin shells and removal of the inflatable mandrel. As noted, the densified and staged outer shells can be retained in the exterior molds for cocuring and bonding to the honeycomb and inner skin.

The inner nose skin is automatically tape wrapped in the same manner as the outer skin shell. Adhesive and honeycomb are applied to the installation of the outer skin and the exterior molds.

The fuselage nose shells are cocured and bonded for 4 hours at 350°F and 40-psig maximum autoclave pressure. The inflatable mandrel is vented to the autoclave pressure when the autoclave pressure equals the bag pressure.

The inflatable mandrel is deflated sufficiently to allow cutting and separating the two fuselage nose sections. After the structures have been cleaned and prepared, the honeycomb/graphite substructure is installed and bonded with thick adhesive to the fuselage nose compartments (Figure 49).

4.3.4 Airplane Assembly

Figure 50 illustrates the composite airplane assembly breakdown, and the relative locations of primary and secondary structural components.

Figure 51 illustrates the joining and assembly of the complete wing box, including installation of ailerons, flaps, fittings, trailing edge, engine pylons, all electrical and hydraulic wiring and tubing, and the leading edges.

Figure 52 illustrates the assembly of the forward constant section fuselage. This includes:

- Location and installation of fuselage frames and floor beams (Figure 48). Drilling and attaching Stations 439 and 947, Figure 53, for joining to nose and aft fuselage.
- Making cutouts for wing box and landing gears, and installing fittings. (Installation of remaining floor panels)

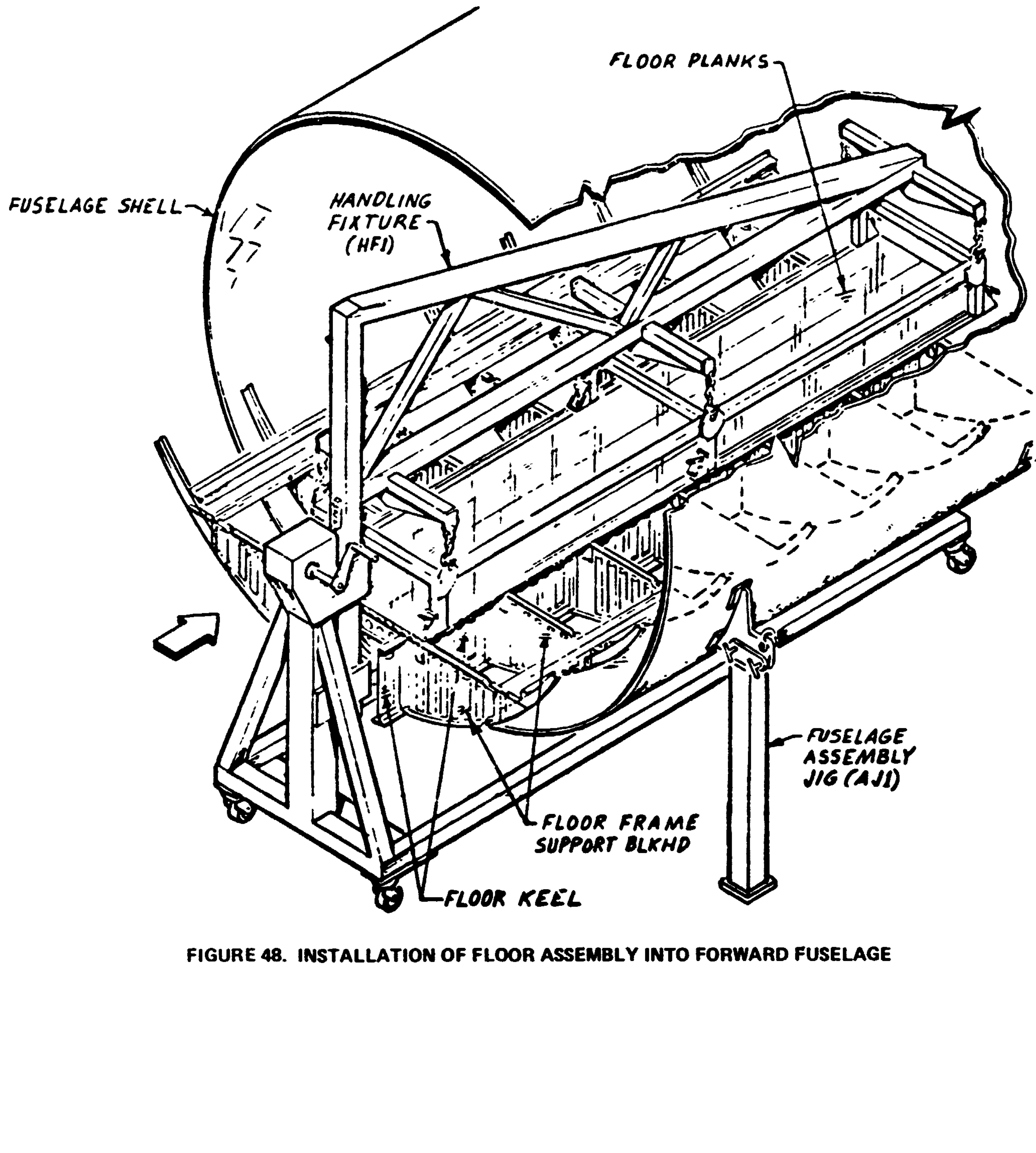


FIGURE 48. INSTALLATION OF FLOOR ASSEMBLY INTO FORWARD FUSELAGE

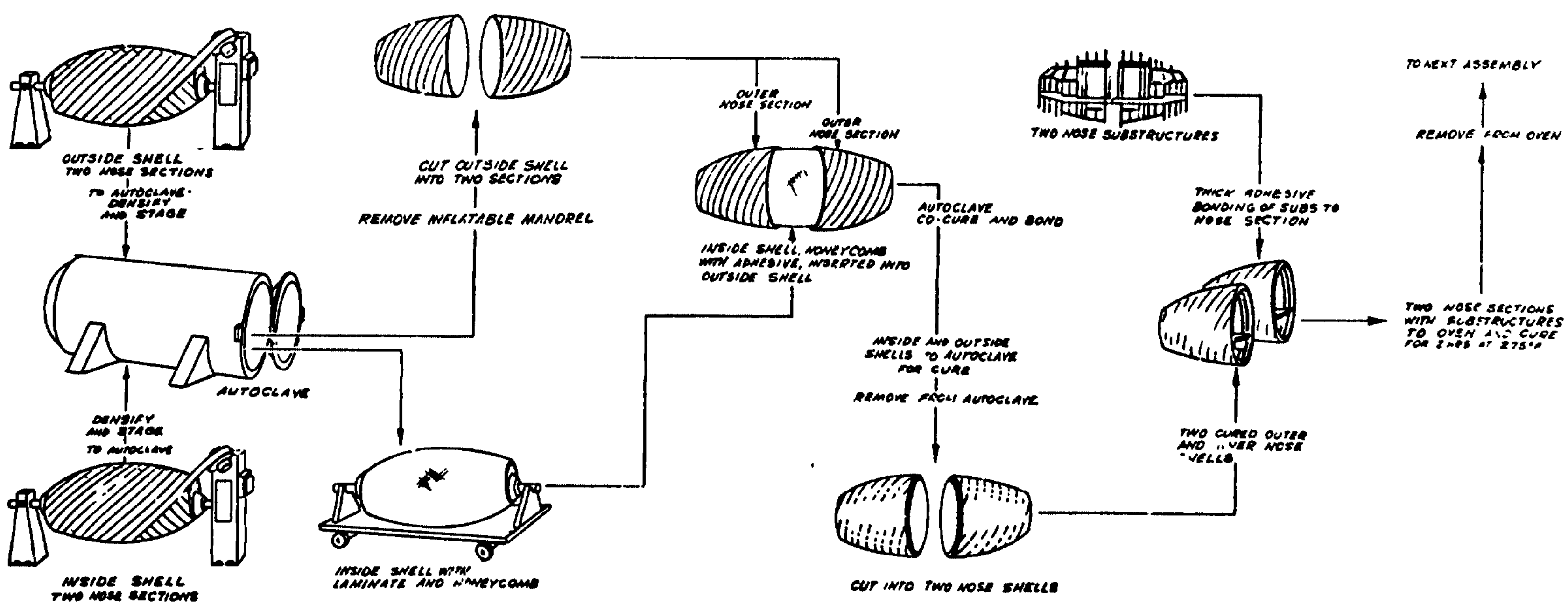


FIGURE 49. FABRICATION OF FUSELAGE NOSE SECTION

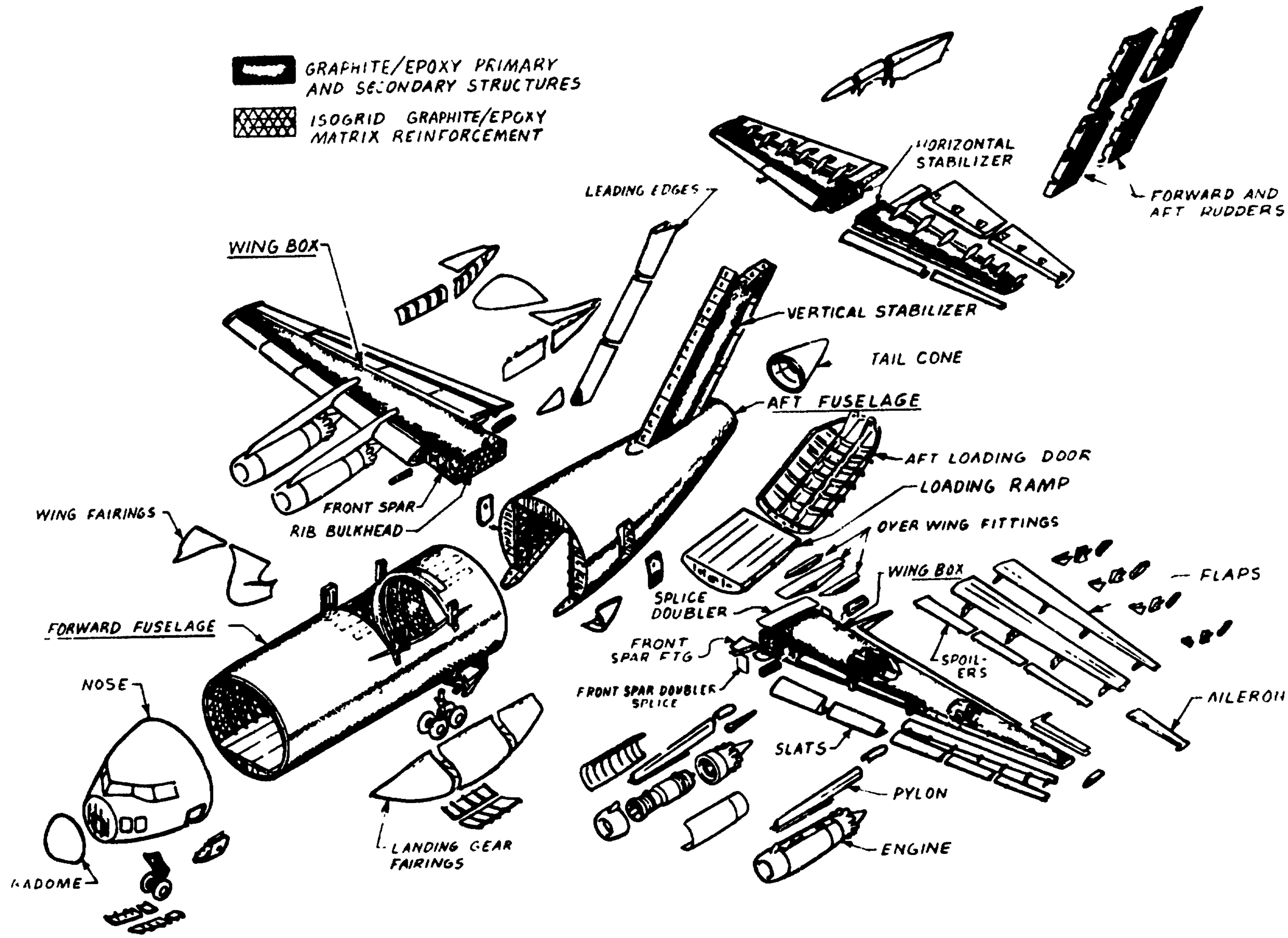
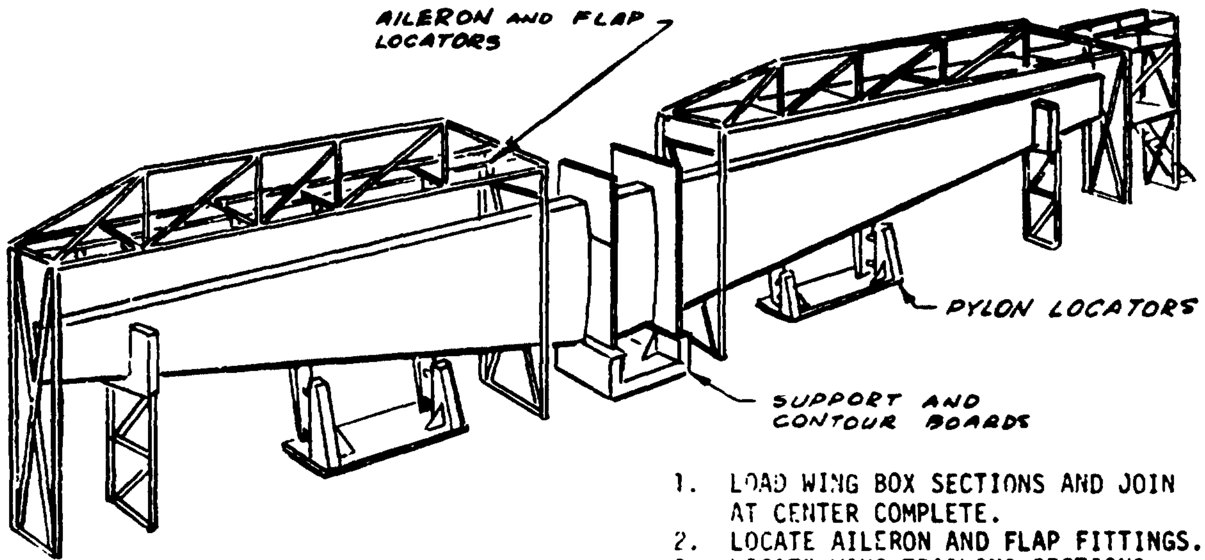
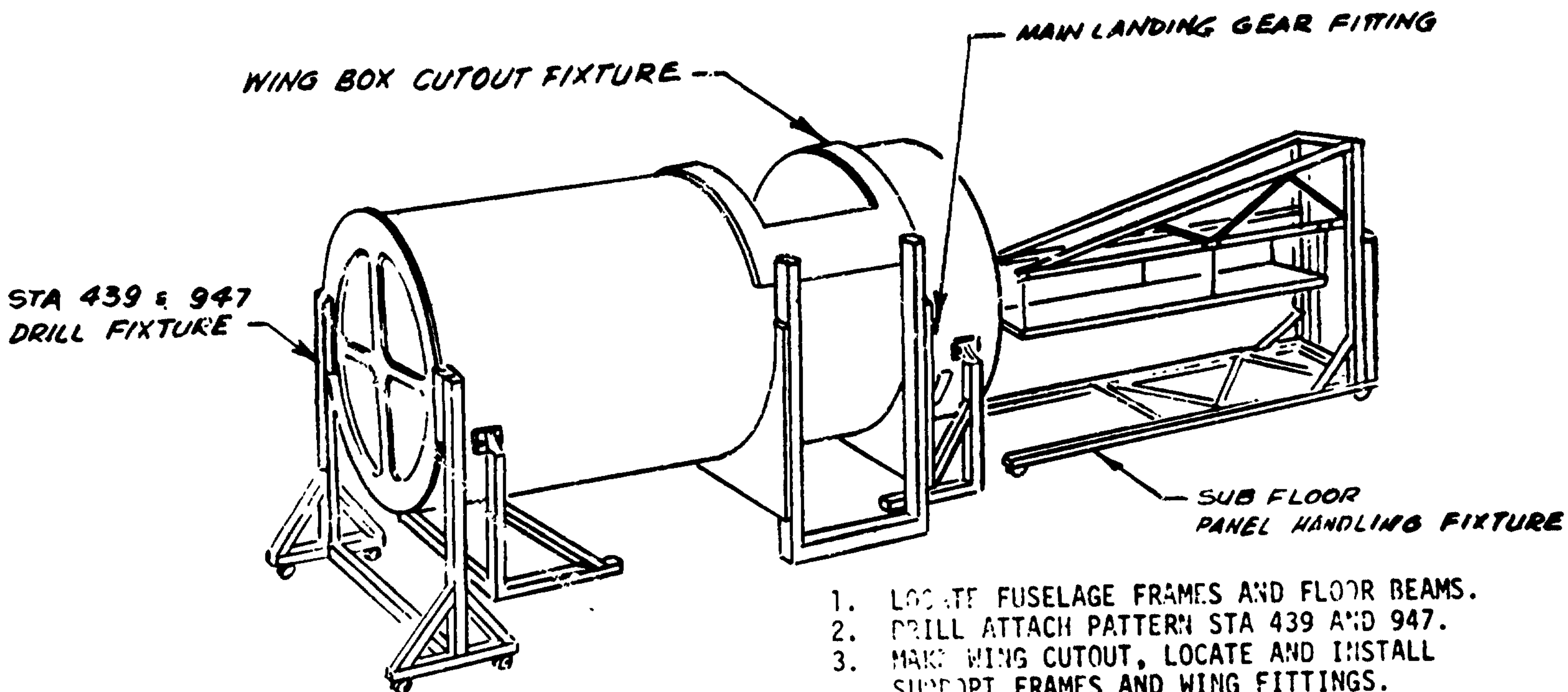


FIGURE 50. COMPOSITE AIRPLANE ASSEMBLY BREAKDOWN



1. LOAD WING BOX SECTIONS AND JOIN AT CENTER COMPLETE.
2. LOCATE AILERON AND FLAP FITTINGS.
3. LOCATE WING TRAILING SECTIONS.
4. LOCATE AND ATTACH ENGINE PYLONS.
5. LOCATE AND FIT WING LEADING EDGE.

FIGURE 51. WING JOINING AND ASSEMBLY



1. LOCATE FUSELAGE FRAMES AND FLOOR BEAMS.
2. DRILL ATTACH PATTERN STA 439 AND 947.
3. MAKE WING CUTOUT, LOCATE AND INSTALL SUPPORT FRAMES AND WING FITTINGS.
4. MAKE LANDING GEAR FITTING CUTOUTS AND LOCATE LANDING GEAR SUPPORT FRAMES AND FITTINGS.
5. INSTALL FLOOR PANELS.

FIGURE 52. FUSELAGE CONSTANT SECTION ASSEMBLY

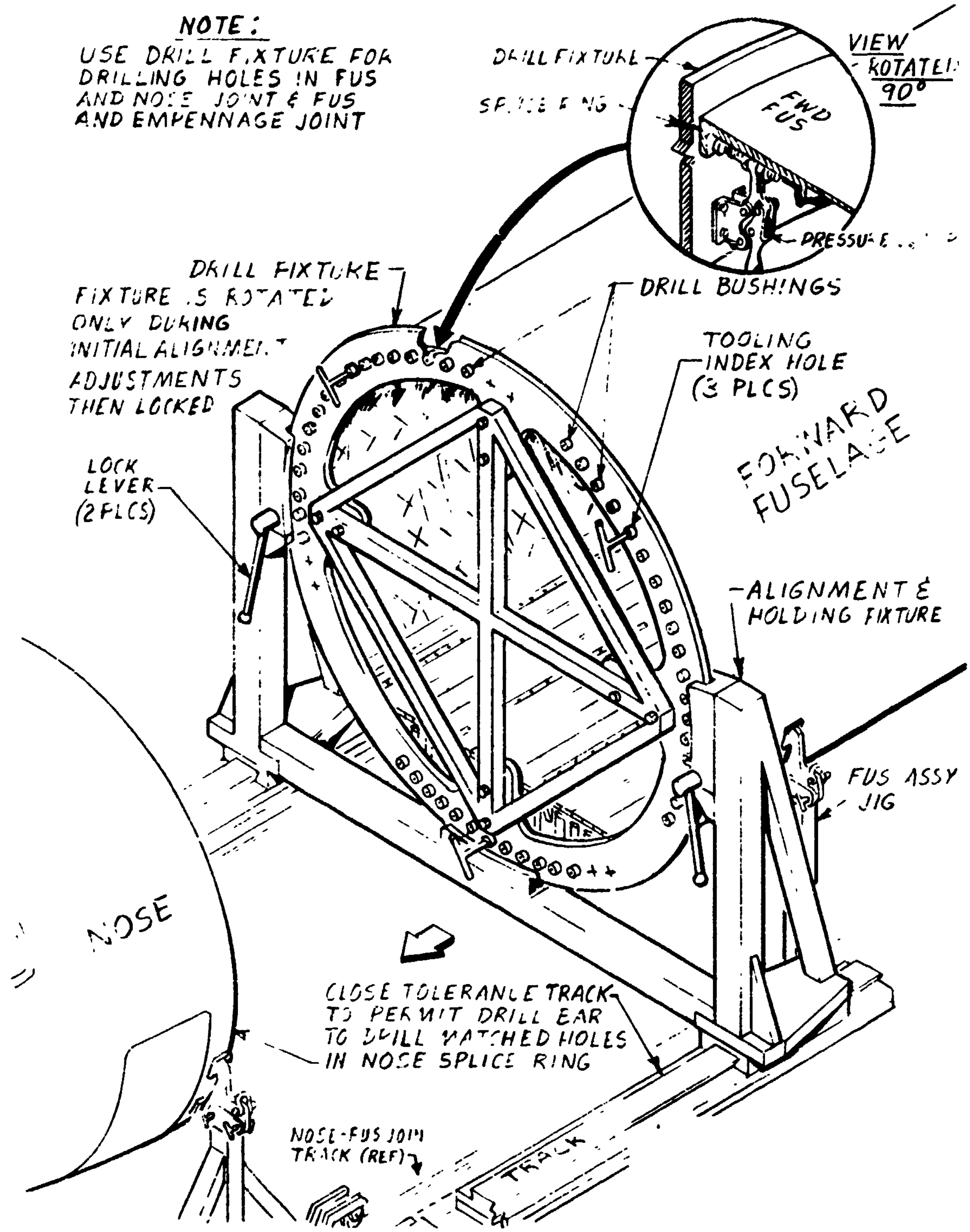


FIGURE 53. FUSELAGE JOINT ASSEMBLY

Figure 54 outlines the sequence for locating frames and beams, cargo door hinges and fittings, cutting out cargo door, and installing door latches. Figure 53 illustrates use of the same drill jig for joining the aft fuselage to the forward fuselage at Station 947, and drilling to attach the fuselage tail cone at Station 1437.

Figure 55 illustrates the aft fuselage/vertical stabilizer assembly fixtures. This includes cutting out opening for installation of fittings and latches for the vertical stabilizer and entrance doors. This is followed by installation, bolting, and torquing of the complete vertical stabilizer assembly, and includes the four rudders, hinge fittings, control push rods, etc.

Figure 56 is a schematic of the assembly tool used for joining the wingbox to the forward fuselage, and installation of the main landing gear.

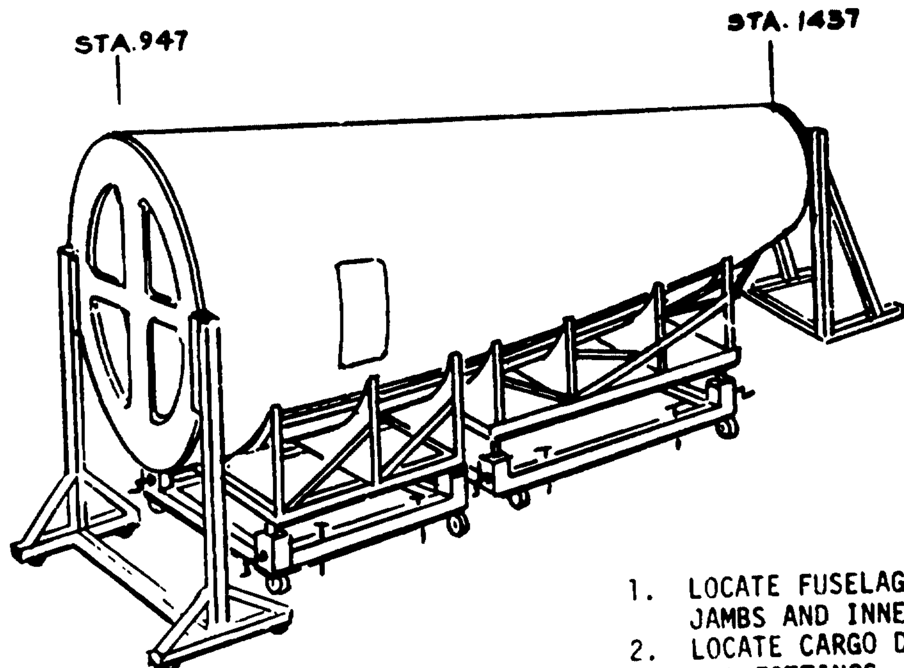
Figures 57, 58, and 59 illustrate the sequence of assembly for the fuselage nose section with floor beams (similar to Figure 48), and thick adhesive bonding of three or four frames at a time in installation fixtures. Figure 58 shows the installation of the nose aft bulkhead door jams and frames, ard also describes locating of the lower main floor beams, the cutting out of doors, and the installation of all hinges and latches.

Figure 59 illustrates the trim, fit, adjustment, and installation of the nose landing gear mounted on a handling fixture. The drill fixture in Figure 53 is again used to drill the hole pattern at Station 439 for mating the nose section to the forward fuselage. Finally, the nose section is trimmed for fit and installation of the radome with hinges and latches.

Figures 60 and 61 illustrate the manufacturing plan for joining the nose and aft fuselage sections to the forward fuselage/wing box assembly. Prior to joining the aft fuselage section, the landing gear and landing gear doors are checked out, removed from dollies, and located on the floor.

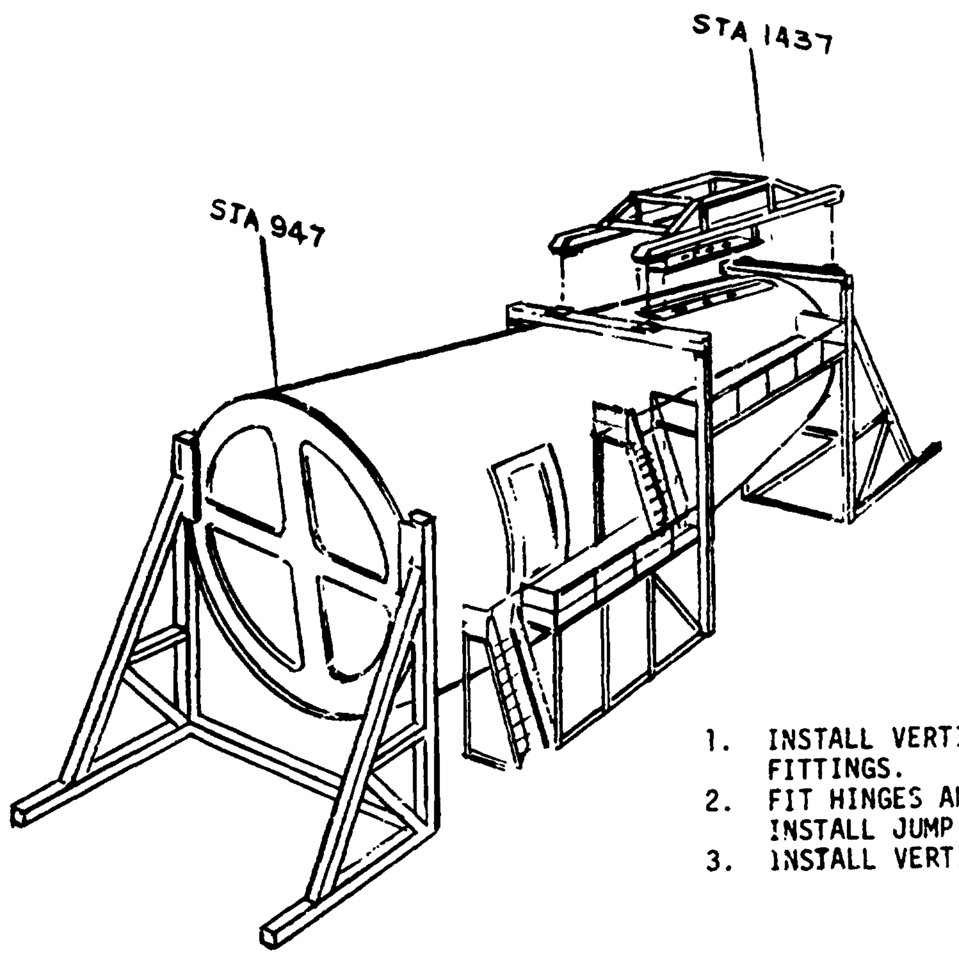
Figure 62 illustrates the final assembly operations for installing the engines, flaps, ailerons, vanes, and leading edges to the wing box. Figure 62 also illustrates installation of elevators, rudders, fairings, and tailcone. The rudders were also called out to be installed in Figure 55, making rudder installation timing optional.

Figure 63 illustrates the composite airplane assembly breakdown with a summary of the appropriate Figures.



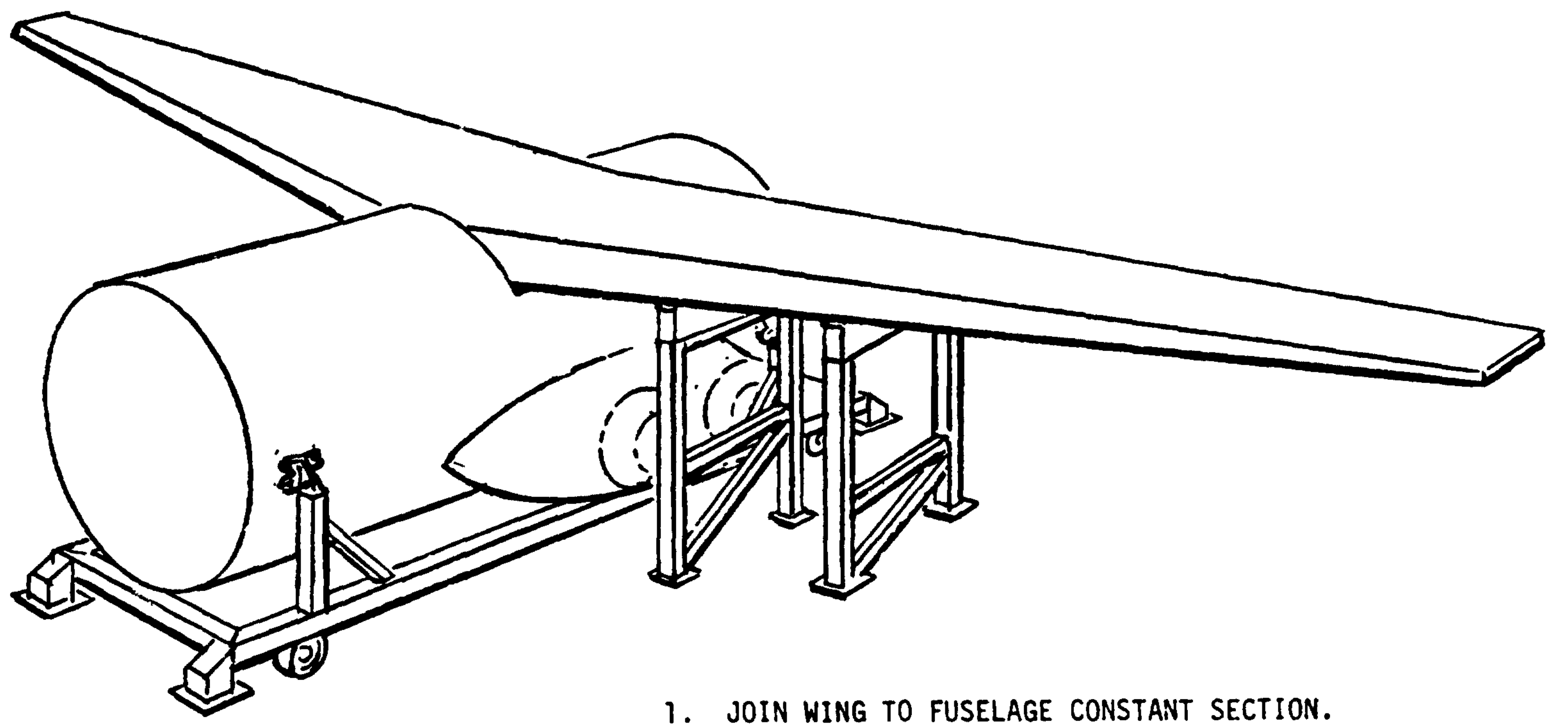
1. LOCATE FUSELAGE FRAMES, BEAMS, DOOR JAMBS AND INNER SPACE FRAME.
2. LOCATE CARGO DOOR AND RAMP HINGES AND FITTINGS.
3. INSTALL CARGO DOOR AND RAMP AND INSTALL LATCHES.
4. DRILL ATTACH PATTERN AT STA 947 AND 1437.

FIGURE 54. AFT FUSELAGE SECTION CARGO DOOR POSITION



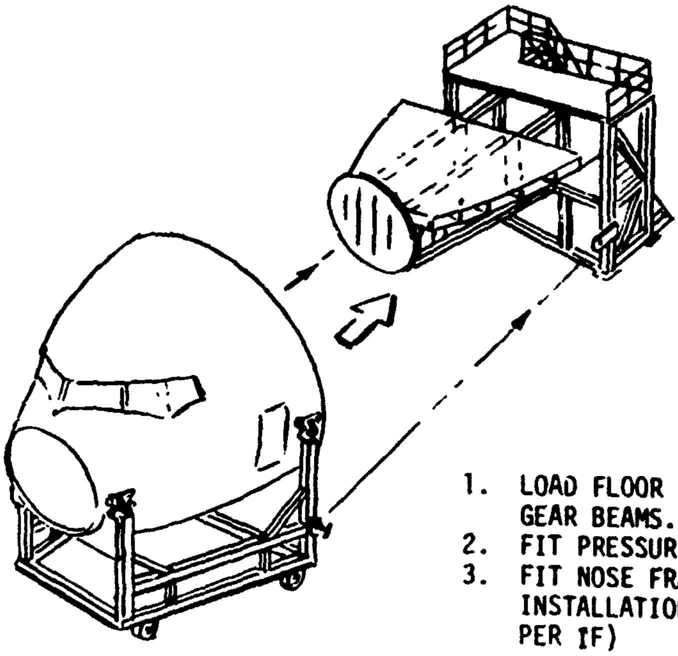
1. INSTALL VERTICAL STABILIZER FITTINGS.
2. FIT HINGES AND LATCHES AND INSTALL JUMP DOORS.
3. INSTALL VERTICAL STABILIZER.

FIGURE 55. AFT FUSELAGE SECTION VERTICAL STABILIZER AND DOOR POSITION



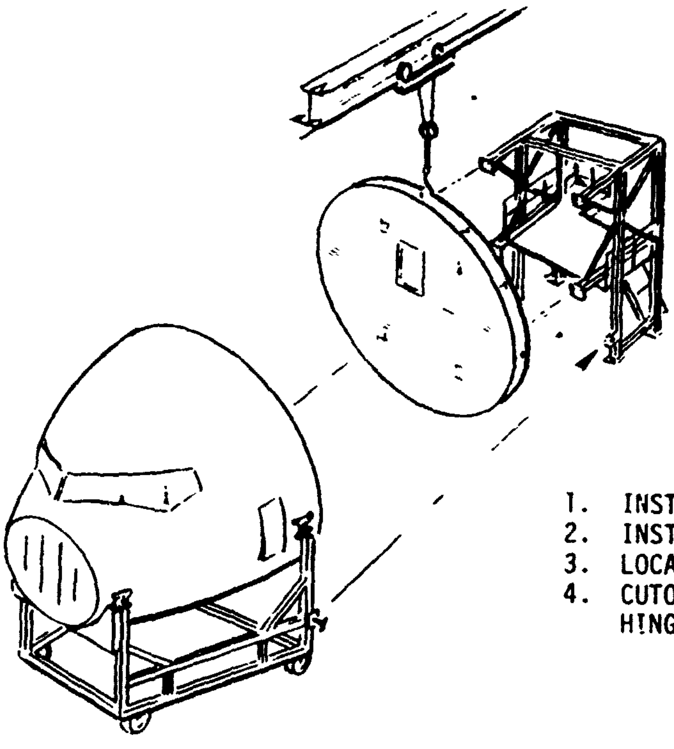
1. JOIN WING TO FUSELAGE CONSTANT SECTION.
2. INSTALL MAIN LANDING GEAR.

FIGURE 56. WING-TO-FUSELAGE JOINING POSITION



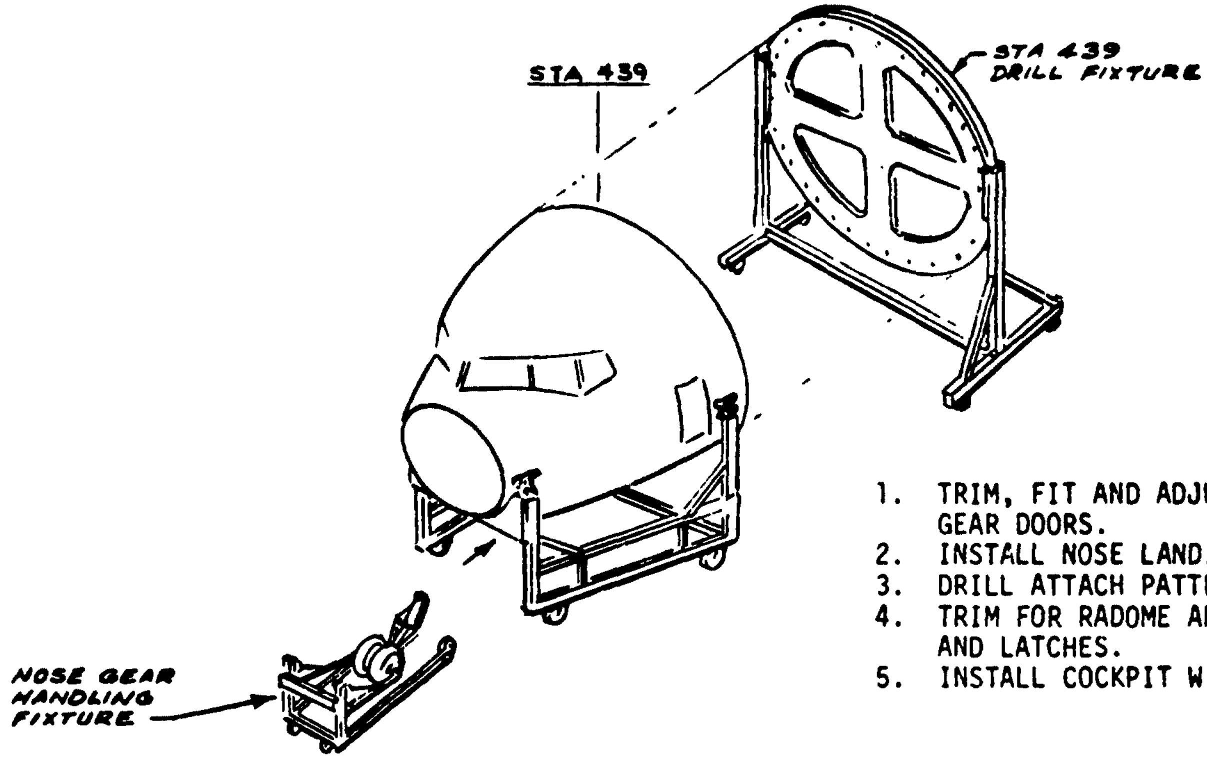
1. LOAD FLOOR BEAMS AND NOSE LANDING GEAR BEAMS.
2. FIT PRESSURE BULKHEAD.
3. FIT NOSE FRAMES AND BOND USING INSTALLATION FIXTURES (3 or 4 FRAMES PER IF)

FIGURE 57. FUSELAGE NOSE ASSEMBLY – SUBSTRUCTURE



1. INSTALL NOSE AFT BULKHEAD
2. INSTALL DOOR JAMB AND AFT NOSE FRAMES.
3. LOCATE MAIN FLOOR SUPPORTS.
4. CUTOUT DOOR OPENING AND FIT DOOR HINGES AND LATCHES.

FIGURE 58. FUSELAGE NOSE ASSEMBLY – ADDITIONAL INSTALLATIONS



1. TRIM, FIT AND ADJUST NOSE LANDING GEAR DOORS.
2. INSTALL NOSE LANDING GEAR.
3. DRILL ATTACH PATTERN AT STA 439.
4. TRIM FOR RADOME AND INSTALL HINGES AND LATCHES.
5. INSTALL COCKPIT WINDOWS.

FIGURE 59. FUSELAGE NOSE ASSEMBLY – COMPLETE

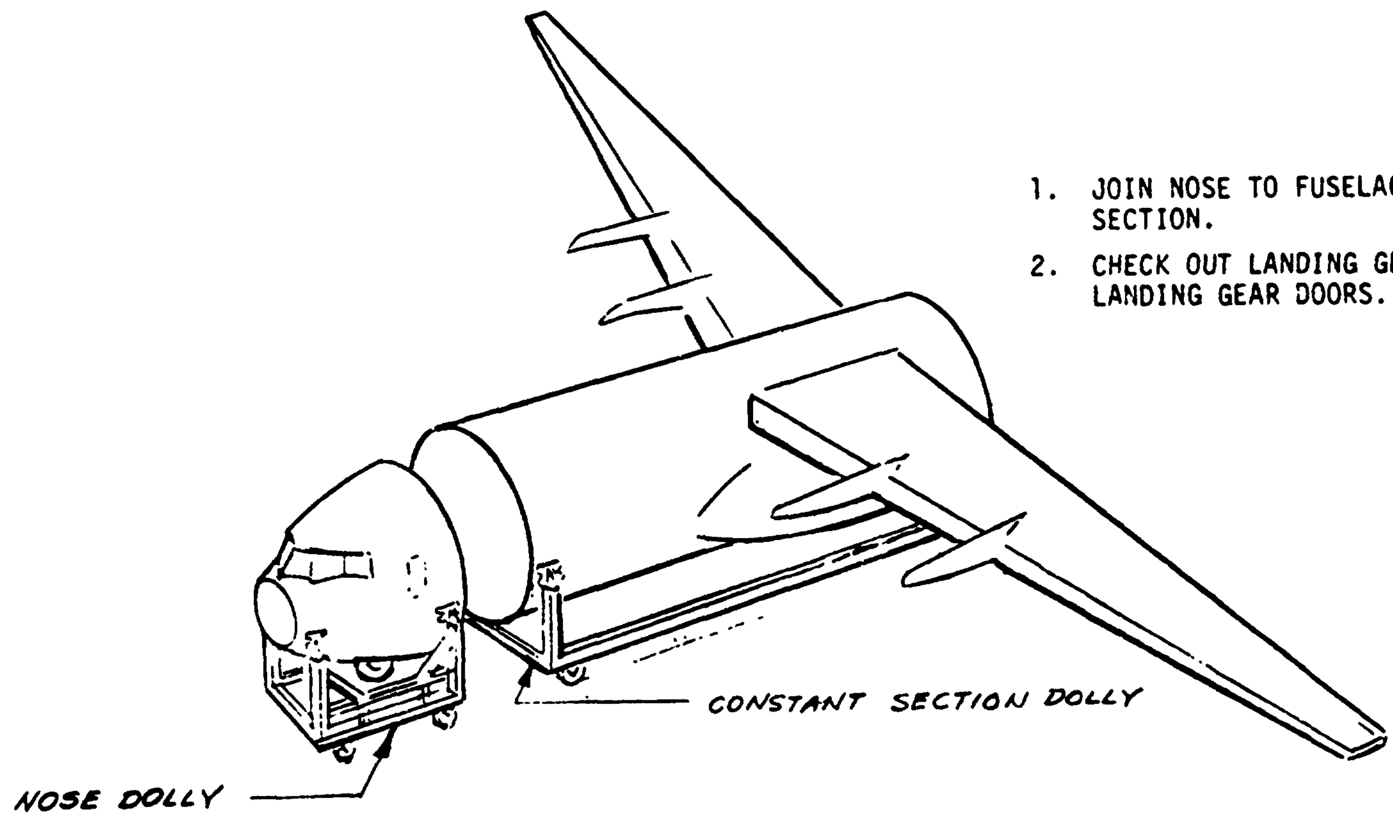
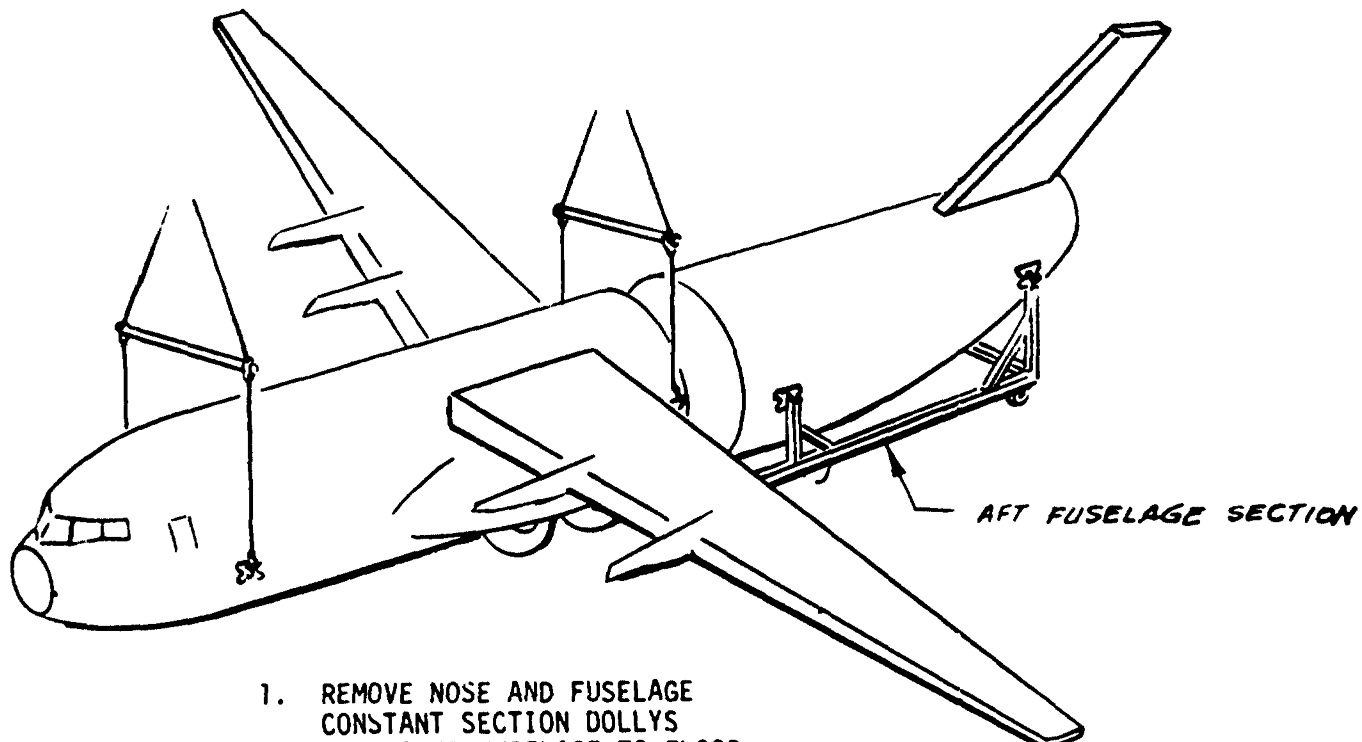
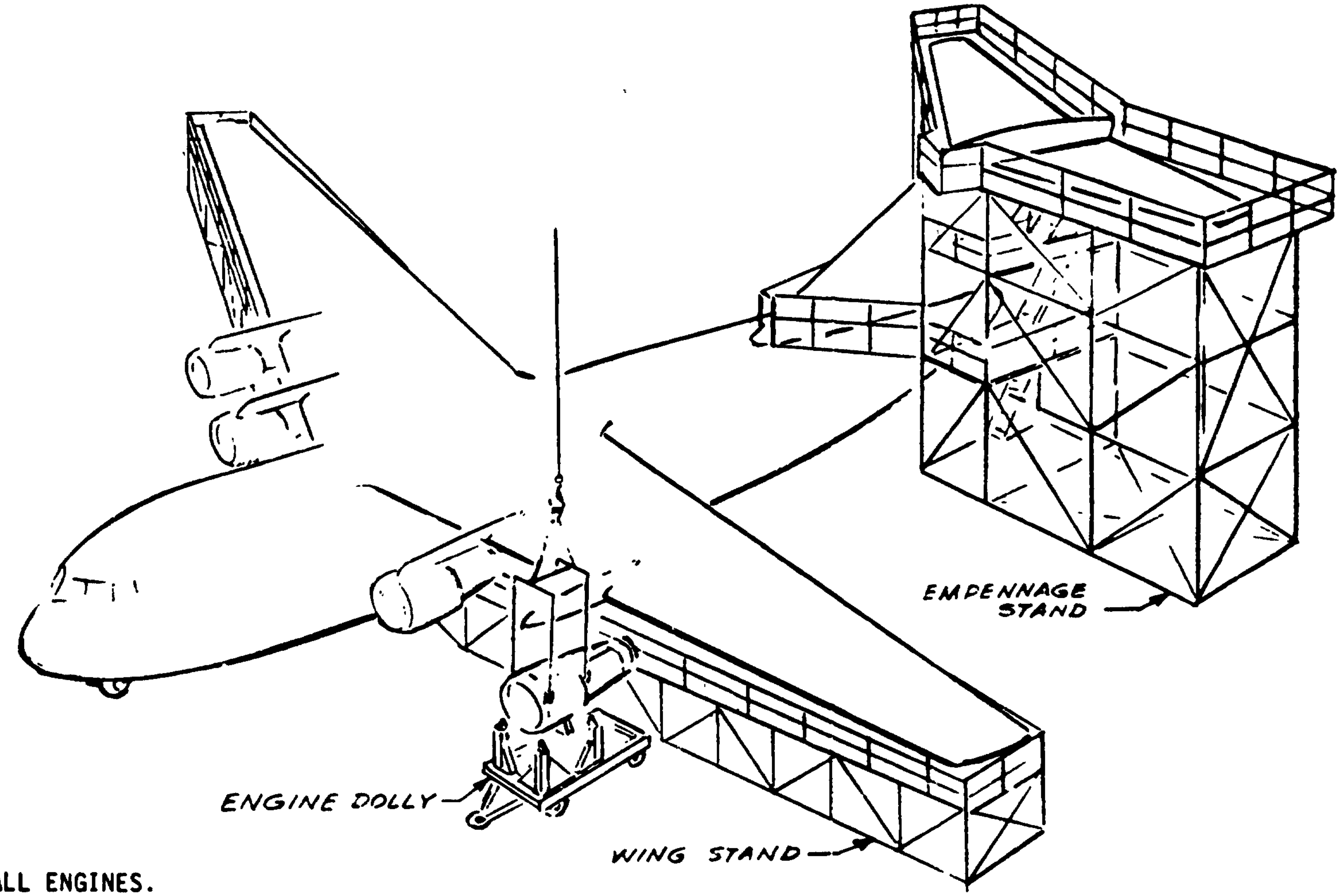


FIGURE 60. NOSE-TO-CONSTANT-SECTION JOINING POSITION



1. REMOVE NOSE AND FUSELAGE CONSTANT SECTION DOLLYS AND LOWER FUSELAGE TO FLOOR LEVEL.
2. JOINT AFT FUSELAGE SECTION

FIGURE 61. FUSELAGE JOINING POSITION



1. INSTALL ENGINES.
2. INSTALL FLAPS, AILERONS, VANES, AND LEADING EDGES.
3. INSTALL HORIZONTAL STABILIZER.
4. INSTALL ELEVATORS, RUDDERS, FAIRING, AND TAILCONE.

FIGURE 62. FINAL INSTALLATION LINE POSITION

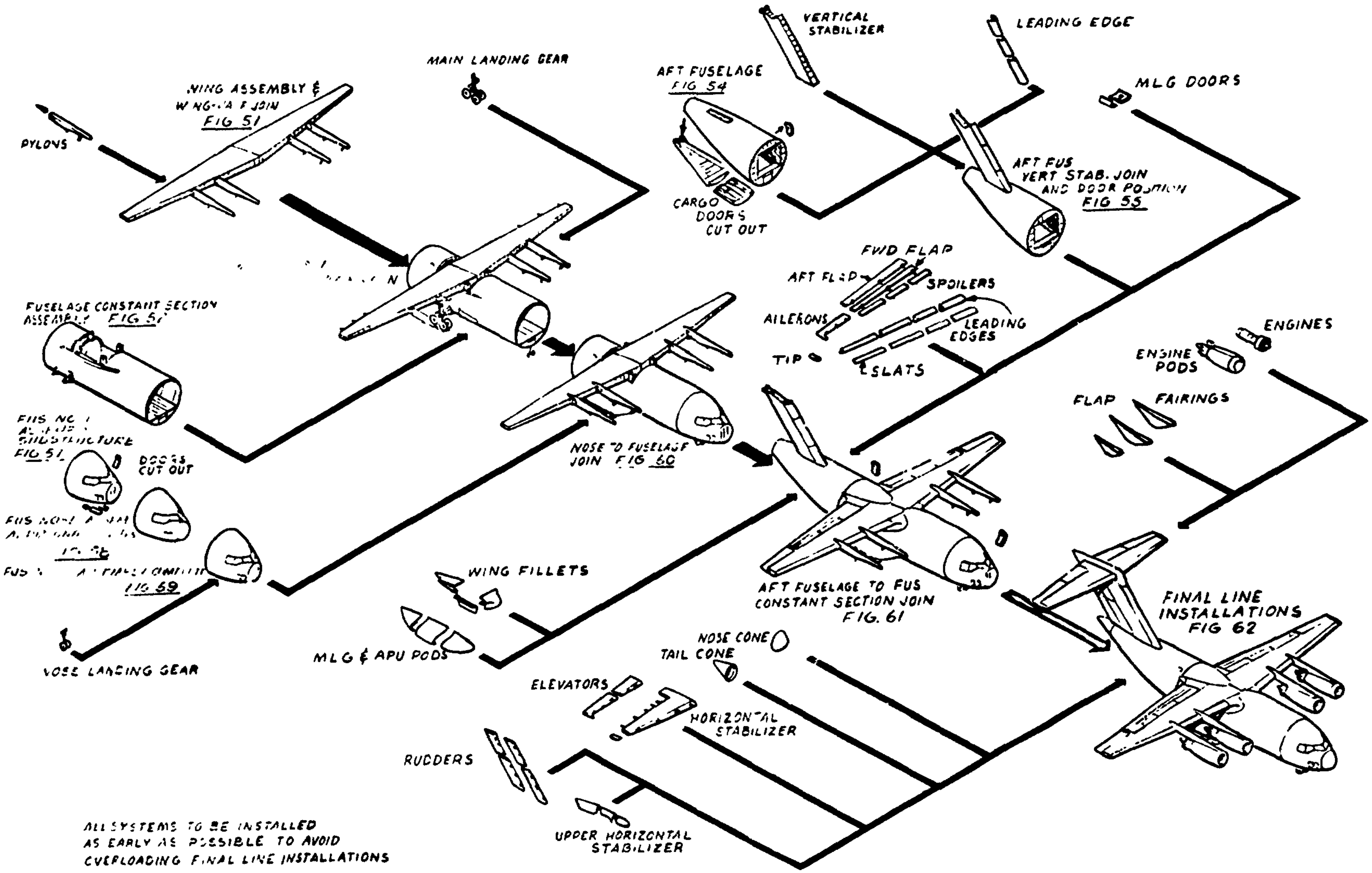


FIGURE 63. COMPOSITE AIRPLANE ASSEMBLY SEQUENCE

SECTION 5
PAYOFF STUDIES

5.1 PERFORMANCE OF COMPOSITE AIRCRAFT

The metal baseline AMST aircraft was resized in order to reflect the reduction in aircraft weight associated with the use of composite materials and to enable assessment of the available performance improvements. Three basic composite aircraft were sized:

1. Unresized Aircraft

2. Resized Aircraft

3. Resized Aircraft with Fixed Thrust

Table 9 is a summary of the performance characteristics of the three composite aircraft as compared to those of the metal baseline aircraft. The definition and performance of the three composite aircraft are discussed in the following subparagraphs.

5.1.1 Unresized Aircraft

This aircraft illustrates the effect of composite materials usage for a fixed geometry. The aircraft has the same external dimensions and engine thrust as the metal baseline aircraft, but portions of the metal structure have been replaced with composites. The reduction in structural weight, 5760 pounds in aircraft empty weight, provides a performance improvement over the metal baseline aircraft. The reduction may be taken as a reduction in field length, an increase in payload, or as an increase in mission radius. Table 10 shows these performance improvement options.

The midpoint TOGW shown in Table 9 corresponds to Options 2 and 3. The reduction in field length for Option 1 occurs at a midpoint TOGW of 143,630 pounds.

5.1.2 Resized Aircraft

The resized aircraft is essentially the minimum weight and cost airplane which meets the same performance requirements as the base aircraft, i.e., the ability to perform a 400-n-mi-radius mission carrying 27,000 pounds of payload with a midpoint hot-day field length of 2000 feet. This aircraft has the same midpoint wing loading (W/S) and thrust-to-weight ratio (T/W) at the mission midpoint as the metal baseline aircraft and therefore the same field length performance. The reduction in structural weight due to the use of composite materials is accompanied by corresponding reductions in wing area and engine size. Engine weight and performance are those of the JT8D-17 engine scaled linearly to the required engine size. Stability and control considerations do not permit the tail areas to decrease quite as fast as wing area.

TABLE 9.
AMST COMPOSITE MATERIALS STUDY
AIRCRAFT CHARACTERISTICS

	METAL BASELINE AIRCRAFT	FULL-SIZE COMPOSITE AIRCRAFT	COMPLETELY RESIZED COMPOSITE AIRCRAFT	PARTIALLY RESIZED COMPOSITE AIRCRAFT
WING AREA (SQ FT)	1740	1740	1607	1545
THRUST/ENGINE (LB)	14,900	14,900	13,760	14,900
MIDPOINT TOGW (LB)	150,000	150,000	138,500	139,890
OEW (LB)	103,240	97,480	92,980	94,000
MIDPOINT WING LOADING (LB/SQ FT)	86	86	86	90.5
MIDPOINT THRUST-TO-WEIGHT RATIO	0.397	0.397	0.397	0.426
DESIGN PAYLOAD (LB)	27,000	32,560	27,000	27,000
DESIGN RADIUS (N MI)	400	400	400	400
MAXIMUM CRUISE (MACH)	0.74	0.74	0.74	0.76
DESIGN FIELD LENGTH (FT)	2000	2000	2000	2000
FERRY RANGE (N MI)	2420	2380	2210	2060

TABLE 10.
UNRESIZED AIRCRAFT PERFORMANCE IMPROVEMENT OPTIONS

OPTION	PAYLOAD (LB)	MISSION RADIUS (N MI)	MIDPOINT FIELD LENGTH (FT)
1	27,000	400	1880
2	27,000	585	2000
3	32,560	400	2000

As shown in Table 9, wing area was reduced from 1740 to 1607 square feet and engine thrust was reduced from 14,900 to 13,760 pounds per engine. The reduction in structural weight amounts to 10,260 pounds, almost twice that for the unresized aircraft. Figure 64 shows the general arrangement, dimensions, and characteristics data for this aircraft.

5.1.3 Resized Aircraft with Fixed Thrust

This aircraft is essentially the practical version of the resized aircraft since, in reality, the JT8D-17 characteristics cannot be "rubberized." Resizing was accomplished by keeping thrust constant at 14,900 pounds per engine and reducing wing area to get the W/S and T/W which provide the same field length, payload, and mission radius capability as the base aircraft. The resulting aircraft, described in Table 9, has a structural weight of just over 1000 pounds higher than the idealized resized aircraft. The higher T/W and W/S of the constant thrust resized aircraft compared to the fully resized aircraft results in a wing area of 1545 square feet compared to 1607 square feet. In addition, there is a small increase in Mach number capability associated with the higher T/W.

The unresized composite aircraft and the resized aircraft with fixed thrust have different wing areas but the same engine size. Figures 65 through 67 show the available performance improvements associated with aircraft sizing at intermediate wing areas with a constant size engine. These plots show performance improvements in terms of increased payload, increased mission radius, and reduced field length.

One performance penalty associated with the use of composite structure is a potential reduction in ferry range as shown in Table 9. This is due to the smaller internal wing volumes associated with smaller wing areas and the thick composite honeycomb skin panels. Fuel capacity could be increased by adding fuel in the landing gear pods or increasing wing thickness ratio.

5.2 WEIGHT ANALYSIS

The weight analysis conducted to determine the weight impact of composite structural designs on the metal baseline AMST aircraft is presented below. Many designs were evaluated and the selected designs are based on reduced manufacturing costs rather than minimum weight.

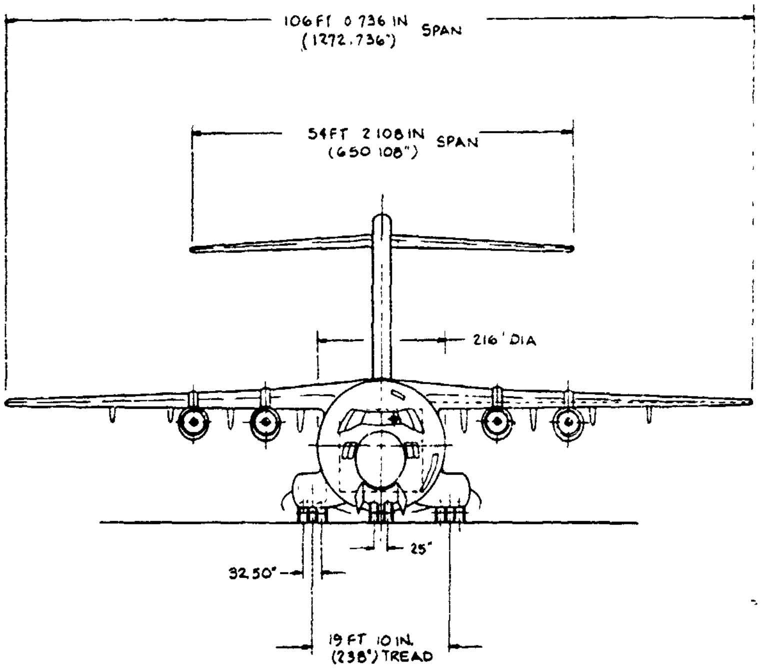
Table 11 presents the group weight summaries for the metal baseline, the unresized composite, and the two composite aircraft resized for the same field length and mission as the metal baseline aircraft. The resized aircraft with fixed thrust keeps the JT8D-17 engine fixed and varies wing loading for constant field length. The completely resized aircraft assumes a "rubberized" engine. Although the JT8D-17 engine cannot be rubberized, this case is provided to indicate the probable maximum payoff for composite usage.

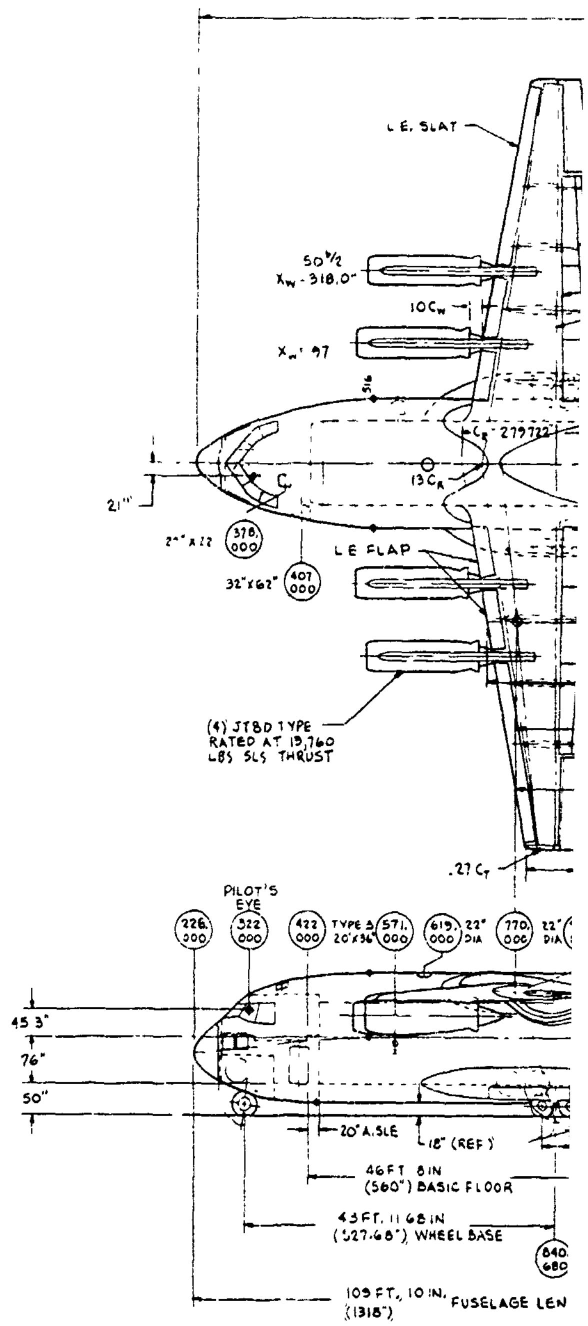
Table 12 details the structural weights for the metal baseline, the unresized, and the resized aircraft. The weights for the composite design of the box structure of the wing and tails, and for the fuselage shell structure are obtained through analysis of the MCE drawings. The aileron, elevator,

CHARACTERISTICS DATA

ITEM	WING (BASIC)	HORIZONTAL TAIL	VERTICAL TAIL
AREA SQ FT	1607	587	417
ASPECT RATIO	7.0	5.0	0.894
TAPER RATIO	0.3	0.45	1.0
SWEEP C/4	5° 53' 6"	4° 46' 12"	4.°
ANHEDRAL	0°	3°	~
THICKNESS % CHORD	13.908% AVERAGE	11%	12%
INCIDENCE	.11 b/2 = 3.437° 323 b/2 = 4.147° 95 b/2 = 1.311°	+5° TO -15°	~
VOLUME RATIO	~	1.323	.1235

CARGO COMPARTMENT SIZE
- 540" LENGTH (EXCLUDES WALKWAY)
- 140" WIDTH
- 136" HEIGHT (MIN)
- 146" HEIGHT (MAX)





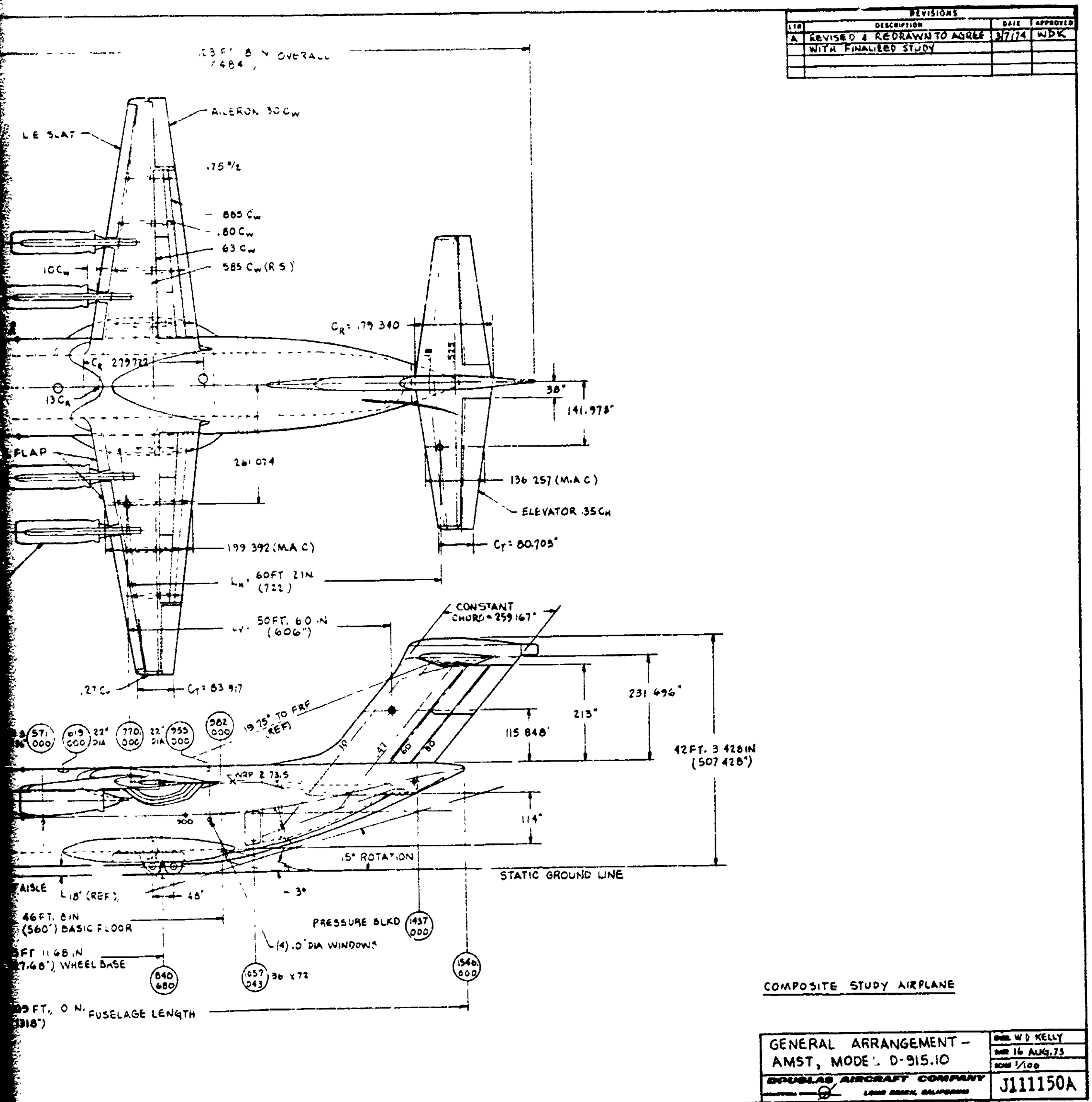


FIGURE 64. RESIZED COMPOSITE AIRCRAFT

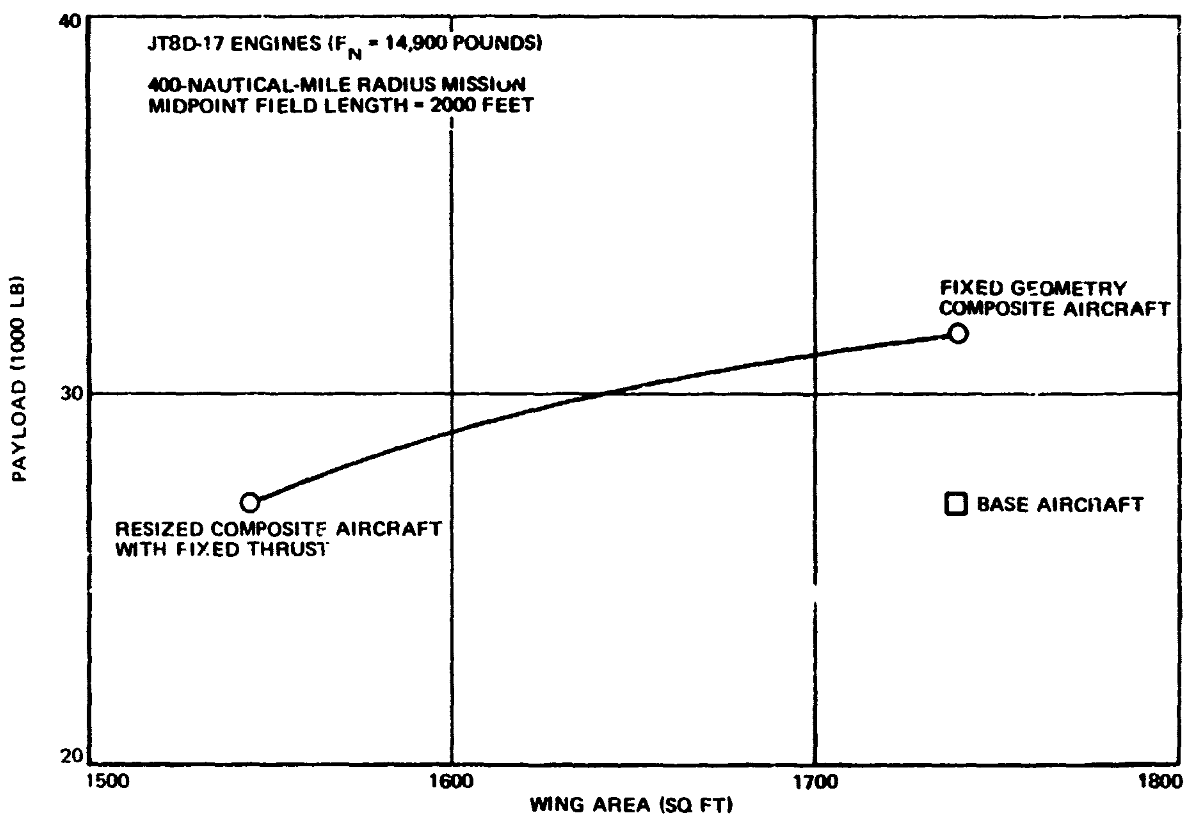


FIGURE 65. AMST COMPOSITE MATERIALS AIRCRAFT MAXIMUM PAYLOAD VS WING AREA

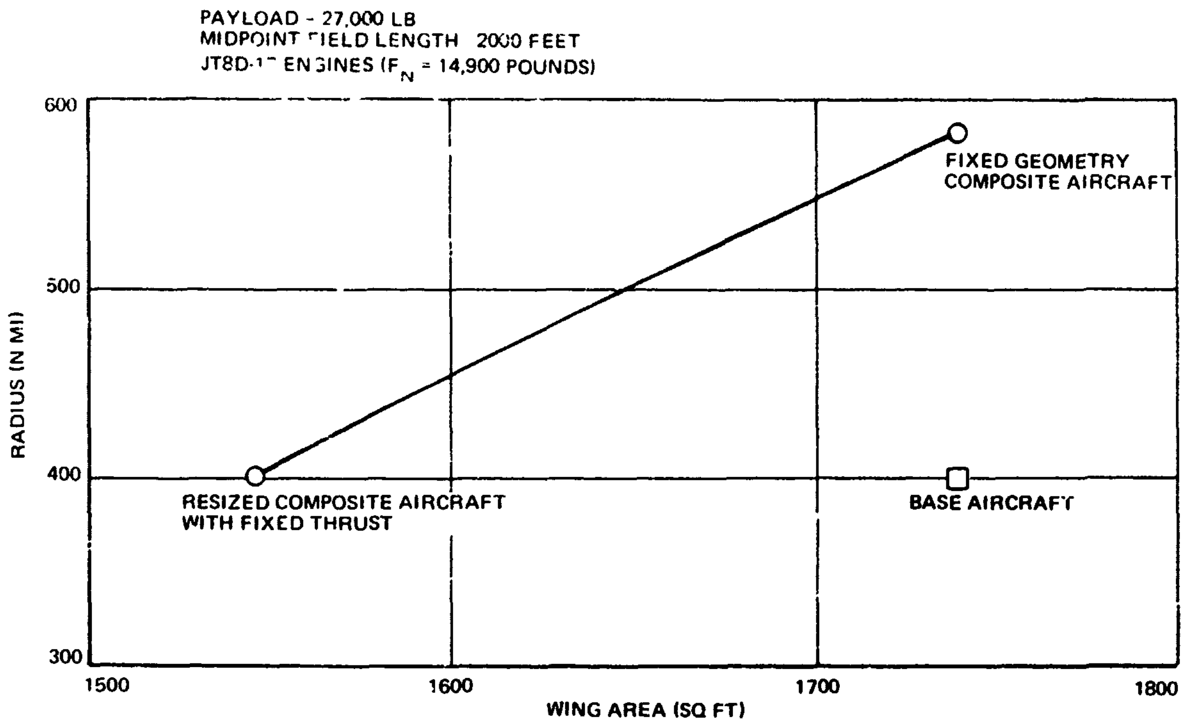


FIGURE 66. AMST COMPOSITE MATERIALS AIRCRAFT MAXIMUM RADIUS VS WING AREA

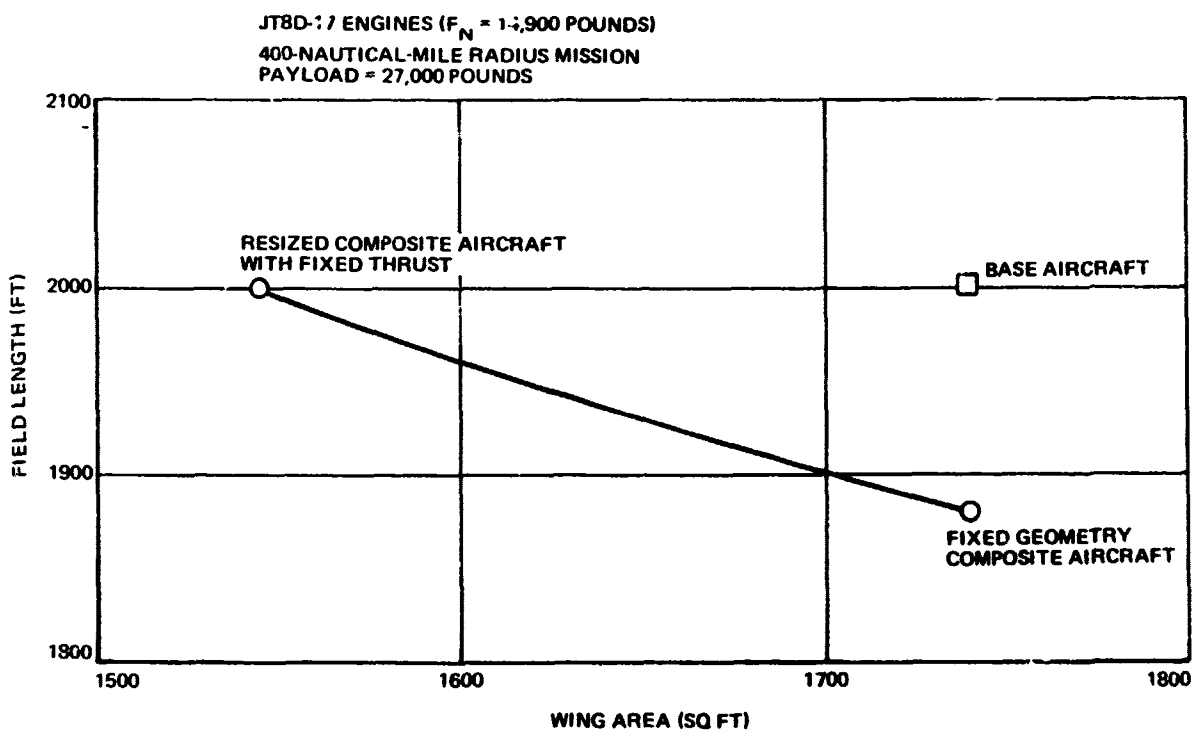


FIGURE 67. AMST COMPOSITE MATERIALS AIRCRAFT MIDPOINT FIELD LENGTH VS WING AREA

TABLE 11
AMST COMPOSITE STUDY GROUP WEIGHT STATEMENT

		A		B		C	
	METAL BASELINE	UNRESIZED COMPOSITE	PERCENT SAVED	RESIZED COMPOSITE	PERCENT SAVED	RESIZED COMPOSITE FIXED ENGINE	PERCENT SAVED
VEHICLE DESCRIPTION							
TAKEOFF WT – STOL (LB)	150,000	150,000		138,500		139,890	
WING AREA (SQ FT)	1,740	1,740		1,607		1,545	
ENGINE DESIGNATION	JT8D-17	JT8D-17		JT8D-17 TYPE		JT8D-17	
ENGINE THRUST (LB/ENG)	14,900	14,900		13,760		14,900	
HORIZ/VERT TAIL AREA (SQ FT)	643/462	643/462		609/449		589/442	
HORIZ/VERT TAIL LENGTH (IN.)	743/616	743/616		743/616		743/616	
HORIZ/VERT TAIL VOLUME	1.323/0.1235	1.323/0.1235		1.4121/0.1352		1.4487/0.1411	
WING LOADING (PSF)	86	86		86		90.5	
THRUST RATIO	0.397	0.397		0.397		0.426	
FUEL FRACTION	0.132	0.170		0.134		0.135	
FUS DIA/LEN (IN.)	216/1318	216/1318		216/1318		216/1318	
WEIGHTS							
WING	18,765	16,369	12.8	14,911	20.5	14,376	23.4
H-TAIL	3,234	2,670	17.4	2,528	21.8	2,446	24.4
V-TAIL	3,460	2,824	18.4	2,744	20.7	2,699	22.0
FUSELAGE	24,367	22,216	8.8	21,953	9.9	21,985	9.8
LANDING GEAR	7,741	7,741		7,147		7,219	
FLIGHT CONTROLS	3,966	3,966		3,773		3,704	
PROPULSION	21,709	21,709		20,048		21,709	
FUEL SYSTEM	768	768		738		724	
APU	966	966		966		966	
INSTRUMENTS	1,453	1,453		1,453		1,453	
HYDRAULICS	1,436	1,436		1,367		1,375	
PNEUMATICS	340	340		340		340	
ELECTRICAL	1,736	1,736		1,736		1,736	
AVIONICS	2,045	2,045		2,045		2,045	
FURNISHINGS	5,497	5,497		5,497		5,497	
AIR CONDITIONING	837	837		837		837	
ICE PROTECTION	254	254		254		254	
HANDLING GEAR	150	150		150		150	
STRUCT WT (NO LG)(1)	53,922	48,175	10.7	45,919	14.8	45,602	15.4
STRUCT WT (WITH LG)(1)	61,663	55,916	9.3	53,066	13.9	52,821	14.3
MFG EMPTY WEIGHT	98,724	92,977	5.8	88,487	10.4	89,515	9.3
OPERATOR'S ITEMS	4,510	4,510		4,493		4,485	
OPERATOR'S EMPTY WT	103,234	97,487		92,890		94,000	
PAYLOAD	27,000	27,000		27,000		27,000	
RETURN SEGMENT FUEL PLUS RESERVES	19,766	25,513(2)		18,520		18,890	
TAKEOFF WT – STOL	150,000	150,000		138,500	7.7	139,890	6.7

(1) INCLUDES NACELLE AND PYLON STRUCTURE (4,096 LB FOR BASELINE)
(2) EXTENDED MISSION

TABLE 12
AMST COMPOSITE STUDY DETAIL STRUCTURAL WEIGHTS

		A		B		C	
	METAL BASELINE	UNRESIZED COMPOSITE	PERCENT SAVED	RESIZED COMPOSITE	PERCENT SAVED	RESIZED COMPOSITE FIXED ENGINE	PERCENT SAVED
WING	(18,765)	(16,369)	(12.8)	(14,911)	(20.5)	(14,376)	(23.4)
BOX STRUCTURE	9,118	6,896	24.4	6,249	31.5	6,065	33.5
AILERON STRUC AND BAL WTS	373	259	30.6	237	36.5	226	39.4
SPOILER STRUCTURE	396	336	15.2	307	22.5	294	25.8
REMAINDER	8,878	8,878	0	8,118	8.6	7,791	12.2
HORIZONTAL TAIL	(3,234)	(2,670)	(17.4)	(2,528)	(21.8)	(2,449)	(24.4)
BOX STRUCTURE	1,749	1,420	18.8	1,344	23.2	1,300	25.7
ELEVATOR STRUCTURE	772	537	30.4	508	34.2	492	36.3
REMAINDER	713	713	0	676	5.2	654	8.3
VERTICAL TAIL	(3,460)	(2,824)	(18.4)	(2,744)	(20.7)	(2,699)	(22.0)
BOX STRUCTURE	1,475	1,112	24.6	1,080	26.8	1,062	28.0
TRAILING EDGE	170	149	12.4	145	14.7	143	15.9
RUDDER STRUCTURE	830	578	30.4	562	32.3	553	33.4
REMAINDER	985	985	0	957	2.8	941	4.5
FUSELAGE	(24,367)	(22,216)	(8.8)	(21,953)	(9.9)	(21,985)	(9.8)
SHELL STRUCTURE	7,897	7,240	8.3	7,240	8.3	7,240	8.3
WING AND MLG SUPPORT	1,409	1,310	7.0	1,166	17.3	1,183	16.0
NLG SUPPORT	49	45	8.2	40	18.4	41	16.3
VERT TAIL SUPPORT	1,277	1,035	19.0	921	27.9	935	26.8
COCKPIT ENCLOSURE	981	930	5.2	930	5.2	930	5.2
NLG PRESSURE PANELS	470	400	14.9	400	14.9	400	14.9
COCKPIT FLOOR AND SUPPORTS	322	275	14.6	275	14.6	275	14.6
CARGO FLOOR AND SUPPORTS	2,892	2,596	10.2	2,596	10.2	2,596	10.2
VEHICLE LOADING CURB	402	402	0	402	0	402	0
NLG DOORS	183	155	15.3	155	15.3	155	15.3
MLG DOORS	747	635	15.0	635	15.0	635	15.0
AFT LOADING DOORS	1,297	1,102	15.0	1,102	15.0	1,102	15.0
RAMP	2,638	2,374	10.0	2,374	10.0	2,374	10.0
PRESSURE BULKHEAD	144	137	4.9	137	4.9	137	4.9
MLG PODS	1,306	1,306	0	1,306	0	1,306	0
RADOME	142	142	0	142	0	142	0
TAILCONE	161	135	16.2	135	16.2	135	16.2
SEALANT	83	83	0	83	0	83	0
COCKPIT LADDER	13	13	0	13	0	13	0
MISC CARGO HANDLING PROVISIONS	125	125	0	125	0	125	0
DOWN VISION WINDOWS	250	250	0	250	0	250	0
TROOP DOORS	212	180	15.1	180	15.1	180	15.1
JUMP DOORS AND DEFLECTOR	854	726	15.0	726	15.0	726	15.0
MISC AND LIFE RAFT DOORS	513	450	12.3	450	12.3	450	12.3
LIGHTNING PROTECTION	0	170	–	170	–	170	–

rudder, and spoiler weight savings are estimated by utilizing experience with the DC-10 upper aft composite rudder design. Control surface supports remain metallic, and the balance weights (ailerons only) are assumed to decrease proportionately to the control surface weight. Estimates of the weight savings in the fuselage secondary structure are based on efforts documented in Reference 1, with emphasis on reduced manufacturing costs.

Tables 13 through 15 present the material breakdowns for the cost analysis. These tabulations are given for the metal baseline, the unresized composite, and the completely resized composite aircraft.

Tables 16 and 17 show the composite usage for the unresized and the resized composite airplanes. While the design may, in a broad sense, be considered an all-composite design, it is only 42 percent by weight composite for the total of the four major structural areas. This is true of the unresized as well as the resized airplanes. The composite weight represents 100-percent graphite epoxy since boron-infiltrated aluminum and glass fiber construction weights are included in the remainder. As is shown in Tables 16 and 17, wing and empennage boxes have the highest composite usage of the component breakdown, being roughly 70-, 72-, and 73-percent graphite epoxy, respectively. The fuselage primary structure has the next highest usage at 57 percent. Tables 16 and 17 also show that an average of approximately 3.2 pounds of composite is used to save 1 pound of unresized airplane weight and approximately 2.3 pounds composite is used per pound saved on the resized airplane. The wing shows the greatest weight saving leverage through resizing. The empennage has a somewhat lesser leverage because of conservative ground rules for tail resizing for stability and control.

5.3 FATIGUE AND STRUCTURAL RELIABILITY ASPECTS OF THE COMPOSITE CONCEPTUAL DESIGN

Filament-reinforced composites have a fundamentally different composition from isotropic metal structures and distinct processing in the form of lamination. Consequently, the mechanical properties exhibited at both the material and structural levels have certain characteristics not shared by metal structures. Some of these represent an improvement, particularly with regard to inplane fatigue loads and reduced material wastage rate during fabrication, while others impose severe limitations which must be designed around. This latter class includes the very poor interlaminar tension and interlaminar shear properties as well as the lack of yielding of the brittle fibers and their significant but limited capability to redistribute loads. How these characteristics have been dealt with for the present design is discussed below. The various structural components will be considered in turn, starting with the fuselage, and proceeding through the wing to the empennage.

The fuselage exterior skin is formed largely from isogrid stiffened non-honeycomb barrel sections. The stiffeners and skin are wound onto an inflatable grooved mandrel and cured together, thereby avoiding the costs and potential unreliability of surface preparations for bonding together in a separate operation. There is a potential problem in keeping the skin and stiffeners together because the only attachment throughout the majority of the structure is the resin matrix. Therefore, in areas of known high-load intensity, and at the manufacturing splices, the stiffeners are supported by

TABLE 13
STRUCTURAL MATERIALS DISTRIBUTION, METAL BASELINE AIRPLANE

	COM-POSITE	GLASS AND FIBER-GLASS	FILLERS, ATTACH., PAINT	AD-HESIVE	7049 ALUM FORG	ALUM NOT FORG	STEEL	TITA-NIUM	ALUM HONEY-COMB	HIGH-DENSITY METAL	BORON-ALUM	TOTAL
WING		(786)	(638)		(3,362)	(10,221)	(681)	(2,930)		(147)		(18,765)
BOX STRUCTURE			543		165	8,410						9,118
AILERON STRUCT AND BAL WT			15		25	186				147		373
SPOILER STRUCTURE			29		44	323						396
REMAINDER		786	51		3,128	1,302	681	2,930				8,878
HORIZONTAL TAIL			(129)		(362)	(2,743)						(3,234)
BOX STRUCTURE			85		55	1,609						1,749
ELEVATOR STRUCTURE			44		67	661						772
REMAINDER					240	473						713
VERTICAL TAIL			(131)		(445)	(2,790)	(94)					(3,460)
BOX STRUCTURE			79		61	1,335						1,475
TRAILING EDGE						170						170
RUDDER STRUCTURE			52		81	697						830
REMAINDER					303	588	94					985
FUSELAGE		(1,315)	(820)		(2,862)	(18,603)	(527)	(240)				(24,367)
PRIMARY STRUCTURE		681	377		2,295	8,884	–	240				12,477
CARGO FLOOR, RAMP, AND SUPPORTS		402	180		320	4,830	200	–				5,932
REMAINDER		232	263		247	4,889	327					5,958
TOTAL		2,101	1,718		7,031	34,357	1,302	3,170		147		49,826

TABLE 14
STRUCTURAL MATERIALS DISTRIBUTION, UNRESIZED COMPOSITE AIRPLANE

	COM-POSITE	GLASS AND FIBER-GLASS	FILLERS, ATTACH., PAINT	AD-HESIVE	ALUM FORG	ALUM NOT FORG	STEEL	TITA-NIUM	ALUM HONEY-COMB	HIGH-DENSITY METAL	BORON-ALUM	TOTAL
WING	(5,179)	(819)	(597)	(291)	(3,293)	(1,605)	(681)	(3,041)	(761)	(102)		(16,369)
BOX STRUCTURE	4,807		512	291	111	303		111	761			6,896
AILERON STRUCT AND BAL WT	96	33	11		17					102		259
SPOILER STRUCTURE	276		23		37							336
REMAINDER		786	51		3,128	1,302	661	2,930				8,878
HORIZONTAL TAIL	(1,354)	(113)	(104)	(87)	(321)	(608)			(83)			(2,670)
BOX STRUCTURE	1,026		67	87	22	135			83			1,420
ELEVATOR STRUCTURE	328	113	37		59							537
REMAINDER					240	473						713
VERTICAL TAIL	(1,250)	(121)	(40)	(102)	(412)	(668)	(94)		(137)			(2,824)
BOX STRUCTURE	815			102	45	13			137			1,112
TRAILING EDGE	82					67						149
RUDDER STRUCTURE	353	121	40		64							578
REMAINDER					303	588	94					985
FUSELAGE	(10,514)	(2,140)	(1,336)	(662)	(2,298)	(1,706)	(608)	(223)	(595)		(2,134)	(22,216)
PRIMARY STRUCTURE	6,499	1,085	646	211	1,721	946	0	223	186			11,517
CARGO FLOOR, RAMP AND SUPPORTS	1,618	402	218	73	320	346	261				2,134	5,372
REMAINDER	2,397	653	472	378	257	414	347		409			5,327
TOTAL	18,297	3,193	2,077	1,142	6,324	4,587	1,383	3,264	1,576	102	2,134	44,079

TABLE 15

STRUCTURAL MATERIALS DISTRIBUTION, RESIZED COMPOSITE AIRPLANE

	COM-POSITE	GLASS AND FIBER-GLASS	FILLERS, ATTACH., PAINT	AD-HESIVE	ALUM FORG	ALUM NOT FORG	STEEL	TITA-NIUM	ALUM HONEY-COMB	HIGH-DENSITY METAL	BORON-ALUM	TOTAL
WING	(4,696)	(749)	(542)	(264)	(3,010)	(1,465)	(623)	(2,779)	(690)	(93)		(14,911)
BOX STRUCTURE	4,356		464	264	100	275		100	690			6,249
AILERON STRUCT AND BAL WT	88	30	10		16					93		237
SPOILER STRUCTURE	252		21		34							307
REMAINDER		719	47		2,860	1,190	623	2,679				8,118
HORIZONTAL TAIL	(1,282)	(107)	(98)	(82)	(304)	(576)			(79)			(2,528)
BOX STRUCTURE	971		63	82	21	128			79			1,344
ELEVATOR STRUCTURE	311	107	35		55							508
REMAINDER					228	448						676
VERTICAL TAIL	(1,215)	(118)	(39)	(99)	(401)	(648)	(91)		(133)			(2,744)
BOX STRUCTURE	792			99	44	12			133			1,080
TRAILING EDGE	80					65						145
RUDDER STRUCTURE	343	118	39		62							562
REMAINDER					295	571	91					957
FUSELAGE	(10,456)	(2,140)	(1,323)	(655)	(2,133)	(1,696)	(608)	(218)	(590)		(2,134)	(21,953)
PRIMARY STRUCTURE	6,441	1,085	633	204	1,556	936	0	218	181			11,254
CARGO FLOOR, RAMP, AND SUPPORTS	1,618	402	218	73	320	346	261				2,134	5,372
REMAINDER	2,397	653	472	378	257	414	347		409			5,327
TOTAL	17,649	3,114	2,002	1,100	5,848	4,385	1,322	2,997	1,492	93	2,134	42,136

TABLE 16
COMPOSITE USAGE – UNRESIZED COMPOSITE AIRPLANE

COMPONENT	COMPOSITE (LB)	TOTAL (LB)	COMPOSITE % OF TOTAL	WEIGHT SAVED (LB)	WEIGHT SAVED PER UNIT WEIGHT OF COMPOSITE
WING	(5,179)	(16,369)	(31.6)	(2,396)	(0.463)
BOX	4,807	6,896	69.7	2,222	0.462
H-TAIL	(1,354)	(2,670)	(50.7)	(564)	(0.417)
BOX	1,026	1,420	72.3	329	0.321
V-TAIL	(1,250)	(2,824)	(44.3)	(636)	(0.509)
BOX	815	1,112	73.3	363	0.445
FUSELAGE	(10,514)	(22,216)	(47.3)	(2,151)	(0.205)
PRIMARY STRUCTURE	6,499	11,517	56.4	960	0.148
CARGO FLOOR, RAMP, SUPPORTS, AND CURB	1,618	5,372	30.1	560	0.346
REMAINDER	2,379	5,327	44.7	631	0.265
TOTAL	18,297	44,079	41.5	5,747	0.314

TABLE 17
COMPOSITE USAGE – RESIZED AIRPLANE

COMPONENT	COMPOSITE (LB)	TOTAL (LB)	COMPOSITE % OF TOTAL	WEIGHT SAVED (LB)	WEIGHT SAVED PER UNIT WEIGHT OF COMPOSITE
WING	(4,696)	(14,911)	(31.5)	(3,854)	(0.821)
BOX ONLY	4,356	6,249	69.7	2,869	0.659
H-TAIL	(1,282)	(2,528)	(50.7)	(706)	(0.551)
BOX ONLY	971	1,344	72.2	405	0.417
V-TAIL	(1,215)	(2,744)	(44.3)	(716)	(0.589)
BOX ONLY	792	1,080	73.3	395	0.499
FUSELAGE	(10,456)	(21,953)	(47.6)	(2,414)	(0.231)
PRIMARY STRUCTURE	6,441	11,254	57.2	1,223	0.190
CARGO FLOOR, RAMP, SUPPORTS, AND CURB	1,618	5,372	30.1	560	0.346
REMAINDER	2,397	5,327	45.0	631	0.263
TOTAL	17,649	42,136	41.9	7,690	0.436

composite angles bonded from the skin to the unidirectional stiffener. This provides fibers to reinforce this joint detail. Based on the available information, all such known problem areas have been covered. However, the detail is not amenable to prediction of either the static strength or fatigue life, so a series of experiments is needed to substantiate the structural integrity of the concept. The design allows use of glass bands in each of the three wrapping directions of the skin to provide triangular crack containment areas.

The intersections of the isogrid stiffeners on the fuselage skin represent a manufacturing task which is justified by the excellent interactive support so obtained. The spreading of the plies as they pass through the intersection eliminates any eccentricities and buildups in thickness. The associated resin-rich area between the crossover and the uniform sections between the intersections does not, in this case, represent a structural deficiency, because there are no filament discontinuities or eccentricities. The small intermediate stiffeners break up the basic triangular panels to effectively reduce the unsupported area acted on by pressure and acoustic loads. These same stiffeners prevent skin panel buckling and thereby protect the structure against the recurrent peeling stresses which would otherwise develop between the stiffener and the buckled skin.

The integrally stiffened central skin barrel is joined to the fore and aft sections at circumferential tension bolt splices with local reinforcement. This minimizes any eccentricity in load path at both of the joints, since the forward section is frame-stiffened honeycomb sandwich with inset fittings at the splice. The fuselage contains a number of major frames to carry load past major cutouts, as at the front of the aft cargo door, and at major load introduction, as with the wing and landing gear frames. In addition, there are equipment support rings to aid in attaching various small concentrated loads to the basic structure so that only distributed loads are applied to the grid. The basic fuselage structure is seen to be sound in concept, but the potential weakness which could give rise to service problems is the great reliance on interlaminar strength. Since it is known that a slow heatup rate during curing of composites leads to poor interlaminar shear strengths, the assurance of proper heatup rates during cure of such a large piece of structure will guide tool and fabrication development.

The aft fuselage is braced internally by a grid of tubes, the intent being to make three small effectively closed torsion tubes (rather than a single large open one) to assist the differential bending action of the sides of the aft fuselage acting as beams. The reason for this added material is the loss of torsional stiffness when the aft cargo door is opened in flight. The termination of this bracing requires two substantial frames at the forward end, one for the reaction of the torque at the front of the door and an additional one to react induced bending loads in the bar structure. The internal truss is fail-safe inasmuch as the removal of a single tube does not convert it to a mechanism. The isogrid stiffening of the skin shares the same desirable feature.

The nose section of the fuselage consists of a honeycomb sandwich shell stiffened by the forward pressure bulkhead, nose gear and floor support structure, and a number of frames. A degree of fail safety is provided by the separate facings on the sandwich and glass rings within the shell laminates.

The fuselage floor is largely boron-epoxy-infiltrated extruded aluminum planks. This kind of construction minimizes the load-introduction problems which are at times associated with the reinforced metals approach because the members are subject to basically bending loads rather than to direct inplane loads. The encapsulation of the boron-epoxy minimizes any possible environmental problems. A potential problem with the use of a metal floor in a basically graphite-epoxy fuselage is the thermal mismatch which could cause a potentially severe fastener problem at the ends of the aluminum sections without proper attention to detail design.

The wing contains solid integrally stiffened front and rear spar shear webs. These are structurally efficient and easy to fabricate which contributes to their reliability. Because of the internal pressure load, the stiffeners will be better located on the inside of the fuel tank rather than on the outside. The reason for this is that, while this means the skin is trying to pull away from the stiffner over most of its length, the peel stresses so induced are far less than those which would be induced locally at the top and bottom of a stiffener located on the outside of the box.

The wing covers have a measure of fail safety due to the sandwich design. Local failure of a sandwich facing transfers load through the core to the other facing. The truss web attachment areas, being solid laminate and containing spanwise relief bands (glass) for fasteners, provide a natural place for limitation of crack width. The panels in themselves are too wide for general crack containment; however, local spanwise glass (or ±45-degree added ply) bands can be incorporated in the panels themselves to break the panel into narrow bands, much as discrete stiffeners on a conventional wing cover operate. The automated layup process for the cover skins uses 1-foot-wide tapes which can be laid with [illegible] between tapes, in the case of the zero-degree plies. Such gaps may be concurrently laid with glass or subsequently filled with ±45-degree "woven" strips, which are 0.010-inch thick. The basic cover ply thickness is double the usual ply thickness to accommodate rapid layup.

The wing covers contain no fastener holes outside the stress-relieved truss crest areas except at the centerline splice, thereby reducing the probability of failure from fatigue stress concentrations. The row of access holes in the wing upper cover is isolated from the rest of the panel by the above failsafe strips in addition to the usual doublers and hole-edge reinforcement.

The above comments also apply to the horizontal stabilizer box since it is of similar design concept.

The vertical stabilizer is a honeycomb sandwich multirib/multispar design which offers considerable redundancy of load path. The spars are terminated at the top of the fuselage and the loads are fed through fittings into the aft fuselage structure. The vertical stabilizer box is terminated at the upper end by the fittings which support the horizontal stabilizer. The honeycomb plates are constant thickness and offer no fabrication problems which would result in inferior quality laminates. No great reliance is placed upon interlaminar strength since the rudder assembly is supported by local metal details. The rudders employ a 2-spar multirib box design covered with thin non-honeycomb skins which can be safely operated in the post-buckled range.

This design is particularly efficient for structural reliability. Letting the skin buckle well prior to ultimate load reduces the amount of composite needed to much below that of a buckle-resistant design. Fatigue tests performed at Douglas showed this construction to be problem free for panel thicknesses representative of those needed for such applications.

The rudder construction is also used for the mass-balanced ailerons, elevators, and spoilers; therefore, identical comments apply to each of these other control surfaces.

SECTION 6

COST ANALYSIS

6.1 COST ANALYSIS OBJECTIVES AND APPROACH

The cost analysis was conducted with two primary objectives: (1) to compare estimated costs for development, acquisition, and operation of an advanced medium STOL transport first utilizing a conventional metal airframe and, alternatively, four configurations utilizing advanced composite airframes, and (2) to indicate the economic feasibility of extensive use of advanced composite materials in the airframe.

Five aircraft configurations were considered in the cost analysis:

- A production AMST baseline metal configuration with JT8D-17 engines
- A dimensionally equivalent configuration making extensive use of composite material (Thornel 300) in the airframe - the wing area and engines were retained identical to the baseline
- A fully resized configuration making extensive use of Thornel 300 composite material and including reduced-scale engine — the lighter weight composite aircraft was resized on the basis of constant thrust loading
- A resized configuration identical to the preceding one only using a pitch-based (low-cost) graphite fiber
- A partially resized composite airframe making extensive use of Thornel 300 composite materials but retaining the baseline JT8D-17 engines.

6.2 COST ANALYSIS METHODOLOGY

The flow diagram for the cost analysis procedure is illustrated in Figure 68. The procedure started with the generation of conceptual designs of the major structural components. During the design process, different materials and fabrication methods were considered for the alternative design concepts. Once a design approach was chosen, a manufacturing cost estimating (MCE) drawing was prepared to define the significant design features in sufficient detail for realistic cost estimating. The MCE drawing formed the basis of all subsequent cost analysis. Manufacturing research and development analyzed the particular component to determine how it would be made in an actual production environment. Tooling and planning personnel developed a fabrication and assembly sequence which, in turn, established the quality control procedures. A bid work sheet was inititated for each structural component to document the manufacturing operations in sequence and then a setup, fabrication, and assembly estimate was prepared for each sequential step as shown in Figure 69.

Six major MCE drawings were prepared to define the structural components of the aircraft investigated in this study (see Appendix C). A total of 1643 bid work sheets were prepared: 799 for the metal baseline and 844 for the composite configurations. Manufacturing estimates (man-hours and material

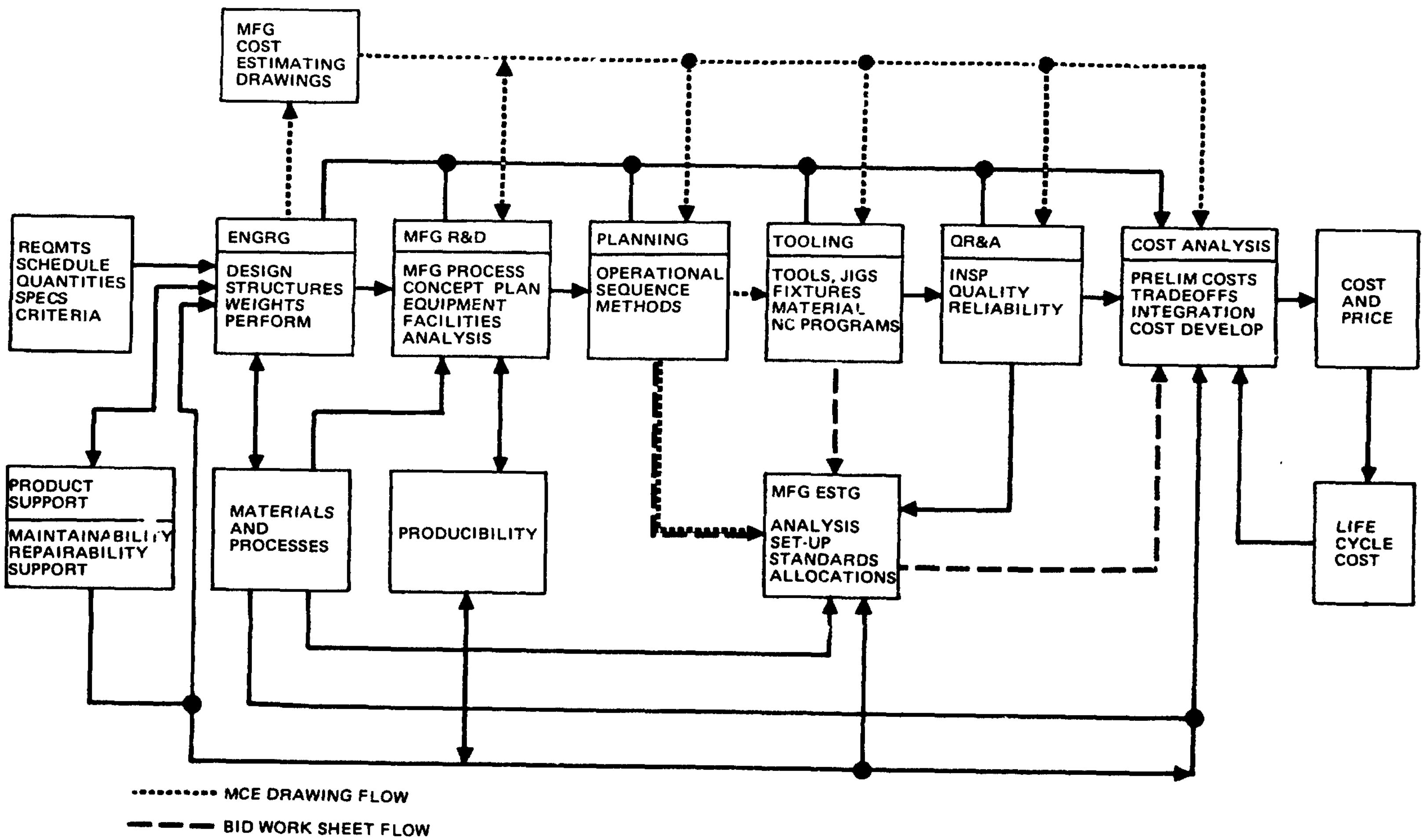


FIGURE 68. COST ANALYSIS INFORMATION FLOW

65 470 (7 64)

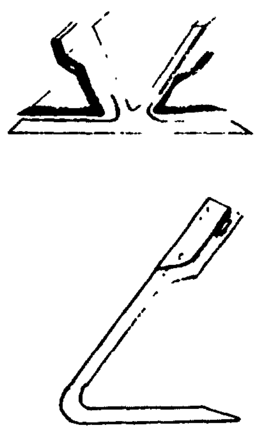
PAGE NO: 1 OF 2

BID WORK SHEET

MAT'L: GRAPHITE/EPOXY	TOTAL NO.REQ.	PART NO: UNASSIGNED	CHG. LET.
SIZE:		PART NAME: CORNER ANGLE - FOR "K" & "T" SECTIONS	
SPEC:		NEXT ASSEM: TO RETAIN HONEYCOMB & SKIN SANDWICH	
		END ITEM:	

PL/	
Q C.	
MAT'L.	
PROC.	
TOOL EST.	
MFG. EST.	

PART ILLUSTRATION: 1.406



UNIT COST SUMMARY		
SET-UP	12.0	HRS
FAB.	5.812	HRS
ASSEM.		HRS
MAT'L.	$	
CYCLE		DAYS

NO.	OPERATION	TOOL	EQUIPMENT	DEPT.	UNIT COST SET-UP	UNIT COST FAB.	UNIT COST ASSEM.	TOOL COST DES.	TOOL COST FAB.
1	SET UP PULTRUSION				8.0	-			
	MACHINE WITH CONTROL		PULTRUSION MACHINE						
	TAPE AND GRAPHITE								
	FEED SPOOLS								
2	SET UP FEED SPOOLS		PULTRUSION MACHINE		4.0				
	FOR 120° ANGLE SECTION								
	PULTRUSION OPERATION								
	FIBER ORIENTATION	%	10 OF PLIES	WIDTH TAPE					
	$\pm 45^{\circ}$	60	16	5.0					
	0°	30	6						
	90°	10	2						
3	FEED ALL TAPES				-	114			
	(PREPREG OR WET LAYUP)		PULTRUSION MACHINE						
	THRU ANGLE SECTION DIE								
	AND PULTRUDE AT THE RATE								
	OF 4 FT. PER MINUTE								
4	CURE TO ADVANCE		PULTRUSION MACHINE		-	-			
	B-STAGE IN PULTRUDER								

DATE:

FIGURE 69. TYPICAL PLANNING BID WORK SHEET

BID WORK SHEET

MAT'L: GRAPHITE/EPOXY	TOTAL NO. REQ.	PART NO: UNASSIGNED	CHG. LET.
SIZE:		PART NAME: CORNER ANGLE - FOR "K" & " " SECTION	
SPEC:		NEXT ASSEM.	
		END ITEM:	

PL.	
Q.C.	
MAT'L.	
PROC.	
TOOL EST.	
MFG. EST.	

PART ILLUSTRATION: 1.406

SEE PAGE 1 FOR ILLUSTRATION.

NO.	OPERATION	TOOL	EQUIPMENT	DEPT.	UNIT COST SET-UP	UNIT COST FAB.	UNIT COST ASSEM.	TOOL COST DES.	TOOL COST FAB.
5	CUT OFF IN 27-FOOT LENGTHS		PULTRUSION MACHINE		-	.024			
6	PLACE IN STAGING AND DENSIFYING DIE	DIE			-	.040			
7	INSERT SILICONE RUBBER PLUG MANDRELS				-	.084			
8	VACUUM BAG				-	5.000			
9	PLACE IN AUTOCLAVE AND DENSIFY AND STAGE AT 250°F AND 100 PSIG FOR 10 MINUTES				-	.040			
10	REMOVE FROM AUTOCLAVE AND REMOVE VACUUM BAG				-	.420			
11	WRAP IN PROTECTIVE BAG AND REFRIGERATE				-	.090			

UNIT COST SUMMARY		
SET-UP		HRS
FAB.		HRS
ASSEM.		HRS
MAT'L.	$	
CYCLE		DAYS

DATE:

FIGURE 69. TYPICAL PLANNING BID WORK SHEET (CONCLUDED)

data) were compiled in accordance with the component definitions and with factors prepared by cost analysis. The final man-hour data were processed by cost analysis and combined with material, instruments, special equipment, avionics, and engine estimates to yield cost and price data of the operating and support portions of life-cycle costs.

The cost information is more typical of trade studies during a real hardware design phase rather than a parametric analysis phase. The estimates reflect the costs of the various processes used to produce the major components and the mix of materials which will be used in a production environment. The mix of materials is particularly important in considering the cost of advanced composite aircraft because the original cost per pound can vary widely among the various material forms. The cost per pound for prepreg tape is quite different from the cost per pound of woven broadgoods.

The cost analysis procedure produces an implicit complexity factor defined by the following general relations:

$$C_F = C_{comp}/C_{BL} \tag{1}$$

and

$$C_F = C_C \times S_F \tag{2}$$

where

C_F = Complexity Factor

C_{comp} = Cost of Composite Component

C_{BL} = Cost of Baseline Component

C_C = Cost Coefficient

S_F = Scale Factor

In contrast, the more commonly used explicit complexity factor rearranges the computational sequence as follows:

$$C_{comp} = C_F \times C_{BL} \tag{3}$$

The implicit complexity factor approach provides a more informative set of information than the explicit complexity factor approach. Further, it specifically addresses the scale factor problem which was pointed out in Reference 11.

The cost analysis procedure reflects the actual material selections, design concepts, and manufacturing processes to be used. The resulting implicit complexity factors for labor and materials are displayed in Tables 18 and 19. These elements indicate substantial variations in the elements of cost. The following data illustrate the variations in results:

Element	Complexity Factor, C_{comp}/C_{BL}		
	Wing Box	Horizontal Stabilizer Box	Fuselage Primary Structure
Manufacturing	0.612	0.928	0.626
Quality Assurance	1.291	1.969	1.395
Material	4.432	11.140	6.391

From the implicit complexity factor table, the cost coefficients can be conveniently calculated by dividing the complexity factor by the scaling factor as indicated by Equation 2. The scaling factor is merely a weight ratio which may be calculated from the weight data listed in Paragraph 6.3.

6.3 ACQUISITION COSTS

The cost analysis procedure described in the previous section used a comparative industrial engineering approach rather than the traditional parametric costing approach using explicit complexity factors. The use of the industrial engineering approach was mandatory to relate the costs to the specific detailed design, fabrication and subassembly, and component processing concepts described in Section 4. The costing technique involved a detailed analysis of the individual MCE drawings with specific attention to:

- Manufacturing steps unique to the alternative design approaches
- Development of the manufacturing plan
- Identification of the tools, equipment, and facilities
- Identification, definition, and application of the material and labor standards and labor allocations
- Identification of the direct labor rates, burden, and factors to establish cost and price.

Finally the comparative approach was calibrated to historical cost performance.

The resulting air vehicle development and production costs for the five study configurations are presented in Tables 20 through 24. All configurations, except the unresized composite aircraft, were developed with constant payload, mission profiles, and landing field requirements. Thus, aircraft performance effectiveness of 4 of the 5 alternative configurations was held constant while the unresized composite aircraft has increased performance. Each program has 5 development aircraft included in the nonrecurring

TABLE 18
IMPLICIT LABOR COMPLEXITY FACTORS
ADVANCED COMPOSITES
BASELINE AIRCRAFT VERSUS RESIZED ADVANCED COMPOSITE AIRCRAFT

AIRCRAFT COMPONENT	MANUFACTURING	QUALITY ASSURANCE	TOOLING	PLANNING
WING				
Wing Box	.612	1.291	.503	.612
Flaps	.899	.900	.914	.899
Ailerons and Balance Weights	.665	1.052	.405	.926
Remainder	.965	1.332	.831	.749
SUBTOTAL	.775	1.109	.750	.836
HORIZONTAL STABILIZER				
Horizontal Box	.928	1.969	1.199	.926
Remainder	.748	1.107	.546	.749
SUBTOTAL	.836	1.522	.782	.836
VERTICAL STABILIZER				
Vertical Box	.648	1.356	.286	.648
Remainder	1.088	1.490	.875	1.087
SUBTOTAL	.871	1.425	.645	.763
FUSELAGE				
Primary Structure	.626	1.395	.811	.626
Remainder	1.036	1.849	.692	1.036
SUBTOTAL	.712	1.492	.771	.712
REMAINDER OF AIRCRAFT[1]	.967	.969	.963	.967
TOTAL	.814	1.184	.812	.809

[1] Includes the following airframe systems:

- o landing gear (less rolling assembly)
- o flight controls
- o propulsion (less engine)
- o fuel system
- o auxiliary power unit
- o instruments
- o hydraulics
- o pneumatics
- o electrical
- o avionics
- o furnishings
- o air conditioning
- o ice protection
- o handling gear

TABLE 19
IMPLICIT MATERIAL COST COMPLEXITY FACTOR
ADVANCED COMPOSITES BASED ON THORNEL 300 FIBER

AIRCRAFT COMPONENT	COMPLEXITY FACTOR
WING	
Wing Box	4.432
Ailerons and Balance Weight	3.792
Spoiler	11.018
Remainder	.936
SUBTOTAL	2.141
HORIZONTAL STABILIZER	
Horizontal Box	11.140
Elevator	7.847
Remainder	.949
SUBTOTAL	7.121
VERTICAL STABILIZER	
Vertical Box	11.389
Trailing Edge	11.265
Rudder	7.781
Remainder	1.051
SUBTOTAL	6.504
FUSELAGE	
Primary Structure	6.391
Cargo Floor, Ramp, Support	6.365
Remainder	8.890
SUBTOTAL	6.840

TABLE 20
DEVELOPMENT AND PRODUCTION COST ESTIMATE
BASELINE METAL AIRCRAFT
300 AIRCRAFT PROGRAM – JANUARY 1973 DOLLARS

RESOURCE ELEMENT	NON-RECURRING	RECURRING	TOTAL
LABOR			
MANUFACTURING	$ 83.004 M	$1261.569 M	$1344.573 M
TOOLING	75.463	194.047	269.510
PLANNING	16.289	125.969	142.258
QUALITY ASSURANCE	19.353	118.882	138.235
ENGINEERING DESIGN	153.835	145.149	298.984
ENGINEERING LABORATORY	45.014	5.068	50.082
FLIGHT TEST	33.761	5.068	38.829
PRODUCT SUPPORT	14.512	5.753	20.265
SUBTOTAL*	$ 441.231 M	$1861.505 M	$2302.736 M
MATERIAL			
MANUFACTURING - RAW MATERIAL & PURCHASED PARTS	$ 17.502 M	$ 239.752 M	$ 257.254 M
EQUIPMENT - INSTRUMENTS AND SPECIAL EQUIPMENT	16.796	303.988	320.784
TOOLING	5.504	11.012	16.516
PRODUCT SUPPORT	12.338	8.130	20.468
FLIGHT TEST	5.266	0.000	5.266
SUBTOTAL**	$ 57.406 M	$ 562.882 M	$ 620.288 M
SUBCONTRACTS			
ENGINES	$ 7.500 M	$ 442.500 M	$ 450.000 M
AVIONICS	2.235	131.865	134.100
SUBTOTAL	$ 9.735 M	$ 574.365 M	$ 584.100 M
TOTAL PRICE	$ 508.372 M	$2998.752 M	$3507.124 M
UNIT PRICE	---	$ 10.165 M	---

*INCLUDES OVERHEAD, G&A, OVERTIME PREMIUM, DIRECT CHARGE, PROFIT
**INCLUDES DIRECT CHARGE AND PROFIT

TABLE 21
DEVELOPMENT AND PRODUCTION COST ESTIMATE
UNRESIZED ADVANCED COMPOSITE AIRCRAFT
300 AIRCRAFT PROGRAM – JANUARY 1973 DOLLARS

RESOURCE ELEMENT	NON-RECURRING	RECURRING	TOTAL
LABOR			
MANUFACTURING	$ 60.086 M	$1080.251 M	$1140.337 M
TOOLING	63.526	163.354	226.880
PLANNING	14.173	109.609	123.782
QUALITY ASSURANCE	22.461	137.977	160.438
ENGINEERING DESIGN	163.651	154.414	318.065
ENGINEERING LABORATORY	53.546	6.021	59.567
FLIGHT TEST	33.761	5.068	38.829
PRODUCT SUPPORT	14.512	5.753	20.265
SUBTOTAL*	$ 425.716 M	$1662.447 M	$2088.163 M
MATERIAL			
MANUFACTURING - RAW MATERIAL & PURCHASED PARTS	$ 37.502 M	$ 513.716 M	$ 551.218 M
EQUIPMENT - INSTRUMENTS AND SPECIAL EQUIPMENT	16.796	303.988	320.784
TOOLING	4.554	9.301	13.855
PRODUCT SUPPORT	12.338	8.130	20.468
FLIGHT TEST	5.266	0.000	5.266
SUBTOTAL**	$ 76.456 M	$ 835.135 M	$ 911.591 M
SUBCONTRACTS			
ENGINES	$ 7.500 M	$ 442.500 M	$ 450.000 M
AVIONICS	2.235	131.865	134.100
SUBTOTAL	$ 9.735 M	$ 574.365 M	$ 584.100 M
TOTAL PRICE	$ 511.907 M	$3071.947 M	$3583.854 M
UNIT PRICE	---	$ 10.413 M	---

*INCLUDES OVERHEAD, G&A, OVERTIME PREMIUM, DIRECT CHARGE, PROFIT
**INCLUDES DIRECT CHARGE AND PROFIT

TABLE 22
DEVELOPMENT AND PRODUCTION COST ESTIMATE
RESIZED ADVANCED COMPOSITE AIRCRAFT
300 AIRCRAFT PROGRAM – JANUARY 1973 DOLLARS

RESOURCE ELEMENT	NON-RECURRING	RECURRING	TOTAL
LABCR			
MANUFACTURING	$ 57.808 M	$1035.756 M	$1093.564 M
TOOLING	60.944	156.714	217.658
PLANNING	13.522	104.575	118.096
QUALITY ASSURANCE	22.615	138.919	161.534
ENGINEERING DESIGN	163.651	154.414	318.065
ENGINEERING LABORATORY	53.546	6.021	59.567
FLIGHT TEST	33.761	5.068	38.829
PRODUCT SUPPORT	14.512	5.753	20.265
SUBTOTAL*	$ 420.359 M	$1607.220 M	$2027.579 M
MATERIAL			
MANUFACTURING - RAW MATERIAL & PURCHASED PARTS	$ 35.901 M	$ 491.797 M	$ 527.698 M
EQUIPMENT - INSTRUMENTS AND SPECIAL EQUIPMENT	16.796	303.988	320.784
TOOLING	4.362	8.910	13.272
PRODUCT SUPPORT	12.338	8.130	20.468
FLIGHT TEST	5.266	0.000	5.266
SUBTOTAL**	$ 74.663 M	$ 812.825 M	$ 887.488 M
SUBCONTRACTS			
ENGINES	$ 6.927 M	$ 408.680 M	$ 415.607 M
AVIONICS	2.235	131.865	134.100
SUBTOTAL	$ 9.162 M	$ 540.545 M	$ 549.707 M
TOTAL PRICE	$ 504.184 M	$2960.590 M	$3464.774 M
UNIT PRICE	---	$ 10.036 M	---

*INCLUDES OVERHEAD, G&A, OVERTIME PREMIUM, DIRECT CHARGE, PROFIT
**INCLUDES DIRECT CHARGE AND PROFIT

TABLE 23
DEVELOPMENT AND PRODUCTION COST ESTIMATE
RESIZED ADVANCED COMPOSITE AIRCRAFT WITH PITCHBASED FIBERS GRAPHITE EPOXY
300 AIRCRAFT PROGRAM – JANUARY 1973 DOLLARS

RESOURCE ELEMENT	NON-RECURRING	RECURRING	TOTAL
LABOR			
MANUFACTURING	$ 57.808 M	$1035.756 M	$1093.564 M
TOOLING	60.944	156.714	217.638
PLANNING	13.522	104.575	118.096
QUALITY ASSURANCE	22.615	138.919	161.534
ENGINEERING DESIGN	163.651	154.414	318.065
ENGINEERING LABORATORY	53.546	6.021	59.567
FLIGHT TEST	33.761	5.068	38.829
PRODUCT SUPPORT	14.512	5.753	20.265
SUBTOTAL*	$ 420.359 M	$1607.220 M	$2027.579 M
MATERIAL			
MANUFACTURING - RAW MATERIAL & PURCHASED PARTS	$ 22.372 M	$ 306.465 M	$ 328.837 M
EQUIPMENT - INSTRUMENTS AND SPECIAL EQUIPMENT	16.796	303.988	320.784
TOOLING	4.362	8.910	13.272
PRODUCT SUPPORT	12.338	8.130	20.468
FLIGHT TEST	5.266	0.000	5.266
SUBTOTAL**	$ 61.134 M	$ 627.493 M	$ 688.627 M
SUBCONTRACTS			
ENGINES	$ 6.927 M	$ 408.680 M	$ 415.607 M
AVIONICS	2.235	131.865	134.100
SUBTOTAL	$ 9.162 M	$ 540.545 M	$ 549.707 M
TOTAL PRICE	$ 490.655 M	$2775.258 M	$3265.913 M
UNIT PRICE	---	$ 9.408 M	---

*INCLUDES OVERHEAD, G&A, OVERTIME PREMIUM, DIRECT CHARGE, PROFIT
**INCLUDES DIRECT CHARGE AND PROFIT

TABLE 24
DEVELOPMENT AND PRODUCTION COST ESTIMATE
RESIZED ADVANCED COMPOSITE AIRCRAFT WITH FIXED ENGINES
300 AIRCRAFT PROGRAM – JANUARY 1973 DOLLARS

RESOURCE ELEMENT	NON-RECURRING	RECURRING	TOTAL
LABOR			
MANUFACTURING	$ 56.633 M	$1014.564 M	$1071.197 M
TOOLING	60.398	155.309	215.707
PLANNING	13.381	103.486	116.867
QUALITY ASSURANCE	22.225	136.530	158.755
ENGINEERING DESIGN	163.651	154.414	318.065
ENGINEERING LABORATORY	53.546	6.021	59.567
FLIGHT TEST	33.761	5.068	38.829
PRODUCT SUPPORT	14.512	5.753	20.265
SUBTOTAL*	$ 418.107 M	$1581.145 M	$1999.252 M
MATERIAL			
MANUFACTURING - RAW MATERIAL & PURCHASED PARTS	$ 35.737 M	$ 489.539 M	$ 525.276 M
EQUIPMENT - INSTRUMENTS AND SPECIAL EQUIPMENT	16.796	303.988	320.784
TOOLING	4.360	8.906	13.266
PRODUCT SUPPORT	12.338	8.130	20.468
FLIGHT TEST	5.266	0.000	5.266
SUBTOTAL**	$ 74.497 M	$ 810.563 M	$ 885.060 M
SUBCONTRACTS			
ENGINES	$ 7.500 M	$ 442.500 M	$ 450.000 M
AVIONICS	2.235	131.865	134.100
SUBTOTAL	$ 9.735 M	$ 574.365 M	$ 584.100 M
TOTAL PRICE	$ 502.339 M	$2966.073 M	$3468.412 M
UNIT PRICE	---	$ 10.054 M	---

*INCLUDES OVERHEAD, G&A, OVERTIME PREMIUM, DIRECT CHARGE, PROFIT
**INCLUDES DIRECT CHARGE AND PROFIT

column and production of 295 production aircraft in the recurring column. All costs are expressed in January 1973 dollars.

6.3.1 Design Differences and Effects

Fundamentally different design concepts were chosen for the primary box structures of baseline and composite configurations as shown in the following tabulation:

	Design Concepts	
	Baseline	Composite
Wing Box	Stringer Stiffened Panels, Conventional Multiple Ribs, Front and Rear Spars	Truss Web
Horizontal Stabilizer Box	Stringer Stiffened Panels, Conventional Multiple Ribs, Front and Rear Spars	Truss Web
Vertical Stabilizer Box	Stringer Stiffened Panels, Conventional Multirib with 2 Main Spars and Auxiliary Spars	Same as Baseline but with sandwich stiffened panels
Fuselage	Skin/Stringer/Frame	Wrapped Isogrid

The differences in the detailed design of the composite components vis-a-vis the conventional designs significantly affect the relative magnitude of the cost elements of the individual components. In some cases, sheer size and part commonality compensated for complexity. In other cases, component size dictated the mix of materials selected and, therefore, the material cost per pound of the component. Each component required a specific mix of fabrication and subassembly activities.

The detailed cost analysis procedures considered both differences in material and labor costs (dollars per pound and hours per pound) and the manufacturing learning rates. The slopes of the learning curves were considered because the distinctions between fabrication and subassembly are less definite than they are using conventional materials. For the composite concepts, an 84-percent learning curve was used.

6.3.2 Labor Hours

The MCE drawings and the individual bid work sheets provided the data for estimating actual (i.e., expected real world) manufacturing hours, materials, and finishing requirements. The individual subassemblies were collected into major components (e.g., wing box, fuselage shell, etc.). Engineering man-hours were estimated by all participating disciplines since sufficient historical data are not presently available. Only minor effects on engineering man-hours were anticipated in some disciplines (e.g., aerodynamics) but others were significantly affected (e.g., structural design, structural mechanics, and the engineering laboratories). Planning was estimated for both baseline and composite cases at 7 percent of manufacturing for the major structural components. Quality assurance was separately estimated on a major component basis depending upon the expected requirements. Tooling hours were estimated consistent with the manufacturing evaluation of the design and tooling concepts, and commonality revealed by the bid work sheets.

The increases projected in design, analysis, and test areas of engineering are based on experience with current and past programs, and stem from the increased complexity of the composite material which is multilayered, anisotropic, notch sensitive, and relatively weak in interlaminar tension. Hence, greater attention to detail is required to assure a design with structural integrity in spite of the fact that fewer drawings may be produced because of fewer and larger parts. Reduction in the drawing release system due to fewer drawings and fewer parts/assemblies could somewhat offset engineering labor increases. To assess that factor would have entailed a detailed estimate of drawing requirements for the airplane.

The cost estimates for the nonresized and the partially resized configurations assumed the same Thornel 300 material used for the fully resized configuration of Table 22. Table 23 contains the cost estimates for the resized composite airplane using the pitch-based fiber material. The detailed manufacturing quality assurance estimates for the basic metal baseline configuration and the resized composite aircraft are presented in Tables 25 and 26. The detailed tooling and planning direct labor hours are shown in Tables 27 and 28 for the baseline and resized configurations. These labor hours and the material cost figures which follow are based on the cumulative average hours per aircraft and the cumulative average dollars per aircraft excluding true nonrecurring costs.

6.3.3 Material Costs

The buy-to-fly ratios, or the material utilization factors, become extremely important as more expensive materials are used. Table 29 shows that conventional buy-to-fly ratios vary between a low of 1.2 for adhesives and aluminum honeycomb to a high of 4.0 for aluminum forgings. On average, four pounds of aluminum forgings are purchased for every pound used in the aircraft, three pounds being removed in detailed machining operations. Conventional material utilization factors represent historical data based on a sample of parts on current Douglas transport aircraft. Material unit costs (Table 29) include internal handling and distribution charges.

The buy-to-fly ratios for the advanced composite material have been estimated after consideration of four loss sources:

1. During the curing process of advanced composites, 10- to 20-percent volatile and resin material is removed.

2. Some material loss is associated with final trim of the cured laminates. Since advanced composites come in several forms, a great deal of shaping may be attained by proper material selection and fabrication techniques. Therefore, the final trim loss will be less than that associated with an aluminum forging.

TABLE 25

BASELINE METAL AIRCRAFT
MANUFACTURING AND QUALITY ASSURANCE LABOR ESTIMATE

AIRCRAFT COMPONENT	DIRECT LABOR HOURS PER AIRCRAFT [1]	
	MANUFACTURING	QUALITY ASSURANCE
WING		
Wing Box	39,498	3,263
Flaps	42,413	3,526
Ailerons and Balance Weights	1,814	155
Remainder	7,412	675
Subtotal	91,137	7,619
HORIZONTAL STABILIZER		
Horizontal Box	6,594	547
Remainder	6,959	589
Subtotal	13,553	1,136
VERTICAL STABILIZER		
Vertical Box	6,488	540
Remainder	6,722	569
Subtotal	13,220	1,109
FUSELAGE		
Primary Structure	61,480	5,212
Remainder	16,416	1,425
Subtotal	77,896	6,637
[2] REMAINDER OF AIRCRAFT	68,520	9,842
TOTAL	264,326	26,343

[1] Cumulative average recurring estimated actual hours

[2] Includes the following airframe systems:

- landing gear (less rolling assembly)
- flight controls
- propulsion (less engines)
- fuel system
- auxiliary power unit
- instruments
- hydraulics
- pneumatics
- electrical
- avionics
- furnishings
- air conditioning
- ice protection
- handling gear

TABLE 26
RESIZED ADVANCED COMPOSITE AIRCRAFT
MANUFACTURING AND QUALITY ASSURANCE LABOR ESTIMATE

AIRCRAFT COMPONENT	DIRECT LABOR HOURS PER AIRCRAFT[1]	
	MANUFACTURING	QUALITY ASSURANCE
WING		
Wing Box	24,169	4,211
Flaps	38,130	3,173
Ailerons and Balance Weights	1,207	163
Remainder	7,152	899
Subtotal	70,658	8,446
HORIZONTAL STABILIZER		
Horizontal Box	6,121	1,077
Remainder	5,207	652
Subtotal	11,328	1,729
VERTICAL STABILIZER		
Vertical Box	4,204	732
Remainder	7,314	848
Subtotal	11,518	1,580
FUSELAGE		
Primary Structure	38,483	7,270
Remainder	17,007	2,635
Subtotal	55,490	9,905
REMAINDER OF AIRCRAFT[2]	66,277	9,532
TOTAL	215,271	31,192

[1] Cumulative average recurring estimated actual hours

[2] Includes the following airframe systems:

- o landing gear (less rolling assembly)
- o flight controls
- o propulsion (less engine)
- o fuel system
- o auxiliary power unit
- o instruments
- o hydraulics
- o pneumatics
- o electrical
- o avionics
- o furnishings
- o air conditioning
- o ice protection
- o handling gear

TABLE 27

BASELINE METAL AIRCRAFT
TOOLING AND PLANNING LABOR ESTIMATE

AIRCRAFT COMPONENT	DIRECT LABOR HOURS PER AIRCRAFT[1]	
	TOOLING	PLANNING
WING		
Wing Box	1,651	2,765
Flaps	2,332	2,969
Ailerons and Balance Weights	220	127
Remainder	2,049	519
Subtotal	6,252	6,380
HORIZONTAL STABILIZER		
Horizontal Box	422	462
Remainder	745	487
Subtotal	1,167	949
VERTICAL STABILIZER		
Vertical Box	455	454
Remainder	711	471
Subtotal	1,166	925
FUSELAGE		
Primary Structure	3,947	4,304
Remainder	1,981	1,149
Subtotal	5,928	5,453
[2]REMAINDER OF AIRCRAFT	5,231	4,798
TOTAL	19,744	18,505

[1]Cumulative average recurring estimated actual hours

[2]Includes the following airframe systems:

- o landing gear (less rolling assembly)
- o flight controls
- o propulsion (less engines)
- o fuel system
- o auxiliary power unit
- o instruments
- o hydraulics
- o pneumatic
- o electrical
- o avionics
- o furnishings
- o air conditioning
- o ice protection
- o handling gear

TABLE 28
RESIZED ADVANCED COMPOSITE AIRCRAFT
TOOLING AND PLANNING LABOR ESTIMATE

AIRCRAFT COMPONENT	DIRECT LABOR HOURS PER AIRCRAFT[1]	
	TOOLING	PLANNING
WING		
Wing Box	830	1,692
Flaps	2,131	2,669
Aileron and Balance Weights	89	84
Remainder	1,703	500
Subtotal	4,753	4,945
HORIZONTAL STABILIZER		
Horizontal Box	506	428
Remainder	407	365
Subtotal	913	793
VERTICAL STABILIZER		
Vertical Box	130	294
Remainder	622	512
Subtotal	752	706
FUSELAGE		
Primary Structure	3,200	2,694
Remainder	1,370	1,190
Subtotal	4,570	3,884
REMAINDER OF AIRCRAFT[2]	5,038	4,639
TOTAL	16,026	14,967

[1] Cumulative average recurring estimated actual hours

[2] Includes the following airframe systems:

- o landing gear (less rolling assembly)
- o flight controls
- o propulsion (less engine)
- o fuel system
- o auxiliary power unit
- o instruments
- o hydraulics
- o pneumatics
- o electrical
- o avionics
- o furnishings
- o air conditioning
- o ice protection
- o handling gear

TABLE 29
MATERIAL UNIT COST

MATERIAL	$/LB JANUARY 1973 DOLLARS	UTILIZATION FACTOR
CONVENTIONAL		
Glass & Fiberglass	2.78	.59
Adhesive	25.66	.83
Aluminum - 7075 Forging	2.46	.25
Aluminum - 7075, 2024 Sheet, Plate, Extrusion	1.64	.81
Aluminum - Honeycomb	8.17	.83
Aluminum - 7049 Forging	2.64	.25
Titanium	9.19	.37
Steel	1.43	.35
Other (Filler, Attachments, Paints, Balance Weights)	4.87	1.00
Aluminum Boron (With 7050 Extrusion)	7.72	.67
ADVANCED COMPOSITES		
Thornel 300 Graphite Epoxy		
12 Inch Tape	38.89	.71
Broadgoods	27.78	.71
Slit Tape	34.44	.71
Cross-Plied Tape	33.33	.71
Pitch Based Fibers Graphite Epoxy		
12 Inch Tape	5.56	.71
Broadgoods	11.11	.71
Slit Tape	12.22	.71
Cross-Plied Tape	16.67	.71

3. During the manufacturing process, samples are taken for quality control purposes. Quality samples are taken from the raw materials and additional samples are taken during the manufacturing process to check process specification compliance. Receiving inspection of a typical small lot might require one-half to one pound for a 200-pound lot and a smaller or similar amount would be required for in-process quality checks depending on size and complexity of part. Larger lot size would not require a proportional increase ir raw material quality samples; however, considerations of storage life, production rate, and statistical sampling techniques would help to determine actual sampling rates. As manufacturing experience is gained, the quality control losses will decrease.

4. In any manufacturing process a number of parts are produced that must be scrapped. The current estimate of scrap loss was based on fiberglass experience As experience is gained with advanced composites, the scrap allowances will undoubtedly decrease.

The final buy-to-fly ratio of 1.4 for composite materials represents the loss ratio anticipated down the production line. During experimental development and early production, higher losses are anticipated.

The raw material cost data were collected from standard procurement sources for 1973. The conventional material cost data and appropriate buy-to-fly ratios were used to calculate the material costs for the four major structural components presented in Tables 30 through 34, for the baseline aircraft. The total raw material and purchased part cost for the baseline aircraft amounts to $9.48 per pound. The structural components account for $5.61 per pound of structure and the remainder amounts to $15.46 per pound of remaining weight. Material remainder consists of those items in the weight breakdown (Table 11) excluding structural items, avionics, and engines, which are separately listed.

The costs and material utilization factors for the advanced composite materials were used along with con ventional materials costs to calculate the material cost for the structural components of the other composite aircraft configurations. The results are summarized in Tables 35 through 38 for the advanced composite aircraft using Thornel 300 material and in Tables 39 through 42 for the advanced composite aircraft using the pitch-based fiber material.

The total raw material and purchased parts cost per pound for the baseline configuration and the completely resized configurations using the Thornel 300 and pitch-based materials are summarized in Tables 30, 43, and 44. This cost per pound information can be summarized as follows:

	Aircraft Cost ($ Per Pound)		
	Structural Components	Remainder	Total
Baseline, Conventional	5.61	15.46	9.48
Resized, Thornel 300	26.51	15.46	21.83
Resized, Pitch-Based Fiber	12.24	15.46	13.60

TABLE 30
BASELINE METAL AIRCRAFT
RAW MATERIAL AND PURCHASED PARTS – SUMMARY[1]

AIRCRAFT COMPONENT	DESIGN COST WEIGHT - LB	COST JANUARY 1973 DOLLARS
WING		
Wing Box	9,118	53,600
Aileron and Balance Weight	373	1,411
Spoiler	396	1,225
Remainder	8,878	112,818
Subtotal	18,765	169,054
HORIZONTAL STABILIZER		
Horizontal Box	1,749	4,696
Elevator	772	2,206
Remainder	713	3,317
Subtotal	3,234	10,219
VERTICAL STABILIZER		
Vertical Box	1,475	4,097
Trailing Edge	170	343
Rudder	830	2,457
Remainder	985	4,546
Subtotal	3,460	11,443
FUSELAGE		
Primary Structure	12,477	55,036
Cargo Floor, Ramp, Supports	5,932	17,632
Remainder	5,958	16,169
Subtotal	24,367	88,837
REMAINDER OF AIRCRAFT[2]	32,229	498,260
TOTAL	82,055	777,813

[1]Cumulative average estimate of 300 quantity

[2]Includes the following airframe systems:

- landing gear (less rolling assembly)
- flight controls
- propulsion (less engine)
- fuel system
- auxiliary power unit
- instruments
- hydraulics
- pneumatic
- electrical
- avionics
- furnishing
- air conditioning
- ice protection
- handling gear

TABLE 31
RAW MATERIAL COST ESTIMATE
BASELINE – 300 AIRCRAFT PROGRAM
WING COMPONENT

MATERIAL CATEGORY	MATERIAL WEIGHT - LB		COST[1] JANUARY 1973 DOLLARS
	DESIGN	PURCHASED	
Fiberglass & Glass	786	1,336	3,714
Adhesive	-	-	-
Aluminum - 7075 Forging	3,197	12,788	31,458
Aluminum - 2024, 7075 Sheet, Plate, Extrusion	1,811	2,228	3,654
Aluminum - 7050 Sheet & Plate (Mostly Sheet)	3,111	3,827	6,812
Aluminum - 7475 Sheet & Plate	1,987	2,444	4,424
Aluminum - 7049 Forging	1,731	6,924	18,279
Aluminum - 7050 Forging	1,746	6,984	21,441
Aluminum - 7050 Extrusion	-	-	-
Boron - Aluminum (With 7050 Extrusion)	-	-	-
Aluminum - Honeycomb	-	-	-
Steel	681	974	2,747
Titanium	2,930	7,911	72,702
Boron	-	-	-
Other (Filler, Attachments, Paint, Balance Weight)	785	785	3,823
Total	18,765	46,201	169,054

[1] Cumulative Average Estimate

TABLE 32
RAW MATERIAL COST ESTIMATE
BASELINE – 300 AIRCRAFT PROGRAM
HORIZONTAL STABILIZER COMPONENT

MATERIAL CATEGORY	MATERIAL WEIGHT - LB		COST[1] JANUARY 1973 DOLLARS
	DESIGN	PURCHASED	
Fiberglass & Glass	-	-	-
Adhesive	-	-	-
Aluminum - 7075 Forging	307	1,228	3,021
Aluminum - 2024, 7075 Sheet, Plate, Extrusion	1,134	1,395	2,288
Aluminum - 7050 Sheet & Plate (Mostly Sheet)	1,073	1,320	2,350
Aluminum - 7475 Sheet & Plate	-	-	-
Aluminum - 7049 Forging	55	220	581
Aluminum - 7050 Forging	-	-	-
Aluminum - 7050 Extrusion	536	659	1,351
Boron - Aluminum (With 7050 Extrusion)	-	-	-
Aluminum - Honeycomb	-	-	-
Steel	-	-	-
Titanium	-	-	-
Boron	-	-	-
Other (Filler, Attachments, Paint, Balance Weight)	129	129	628
Total	3,234	4,951	10,219

[1] Cumulative Average Estimate

TABLE 33
RAW MATERIAL COST ESTIMATE
BASELINE – 300 AIRCRAFT PROGRAM
VERTICAL STABILIZER COMPONENT

MATERIAL CATEGORY	MATERIAL WEIGHT - LB		COST[1] JANUARY 1973 DOLLARS
	DESIGN	PURCHASED	
Fiberglass & Glass	-	-	-
Adhesive	-	-	-
Aluminum - 7075 Forging	384	1,536	3,779
Aluminum - 2024, 7075 Sheet, Plate, Extrusion	1,455	1,790	2,936
Aluminum - 7050 Sheet & Plate (Mostly Sheet)	890	1,094	1,947
Aluminum - 7475 Sheet & Plate	-	-	-
Aluminum - 7049 Forging	61	244	644
Aluminum - 7050 Forging	-	-	-
Aluminum - 7050 Extrusion	445	547	1,121
Boron - Aluminum (With 7050 Extrusion)	-	-	-
Aluminum - Honeycomb	-	-	-
Steel	94	134	378
Titanium	-	-	-
Boron	-	-	-
Other (Filler, Attachments, Paint, Balance Weight)	131	131	638
Total	3,460	5,476	11,443

[1] Cumulative Average Estimate

TABLE 34
RAW MATERIAL COST ESTIMATE
BASELINE – 300 AIRCRAFT PROGRAM
FUSELAGE COMPONENT

MATERIAL CATEGORY	MATERIAL WEIGHT - LB		COST[1] JANUARY 1973 DOLLARS
	DESIGN	PURCHASED	
Fiberglass & Glass	1,315	2,236	6,216
Adhesive	-	-	-
Aluminum - 7075 Forging	-	-	-
Aluminum - 2024, 7075 Sheet, Plate, Extrusion	12,679	15,595	25,576
Aluminum - 7050 Sheet & Plate (Mostly Sheet)	-	-	-
Aluminum - 7475 Sheet & Plate	644	792	1,434
Aluminum - 7049 Forging	2,862	11,448	30.223
Aluminum - 7050 Forging	-	-	-
Aluminum - 7050 Extrusion	5,280	6,494	13,313
Boron-Infiltrated Aluminum (With 7050 Extrusion)	-	-	-
Aluminum - Honeycomb	-	-	-
Steel	527	754	2,126
Titanium	240	648	5,955
Boron	-	-	-
Other (Filler, Attachments, Paint, Balance Weight)	820	820	3,994
Total	24,367	38,787	88,837

[1] Cumulative Average Estimate

TABLE 35
RESIZED ADVANCED COMPOSITE AIRCRAFT
RAW MATERIAL ESTIMATE – WING[1]

Material Category	Material Weight-Lb		Cost January 1973 Dollars
	Design	Purchased	
Conventional			
Glass & Fiberglass	749	1,273	3,539
Adhesive	264	317	8,134
Aluminum - 7049 Forging	-	-	-
Aluminum - 7075 Forging	3,010	12,040	29,618
Aluminum - 2024, 7075	1,465	1,802	2,955
Sheet, Plate, Extrusion	1,465	1,802	
Steel	623	1,757	4,955
Titanium	2,779	7,503	68,952
Aluminum Honeycomb	690	828	6,765
Boron Aluminum	-	-	-
Other (Filler, Attachments, Paint, Balance Weights)	635	635	3,093
Subtotal	10,215	26,155	128,011
Advanced Composite			
Thornel 300 Graphite Epoxy			
12-Inch Tape	2,272	3,181	123,709
Broadgoods	473	662	18,390
Slit Tape	566	792	27,276
Cross-plied Tape	1,385	1,939	64,627
Subtotal	4,696	6,574	234,002
Total	14,911	32,729	362,013

[1]Cumulative average estimate of 300 quantity

TABLE 36
RESIZED ADVANCED COMPOSITE AIRCRAFT
RAW MATERIAL ESTIMATE – HORIZONTAL STABILIZER[1]

Material Category	Material Weight-Lb		Cost January 1973 Dollars
	Design	Purchased	
Conventional			
Glass & Fiberglass	107	182	506
Adhesive	82	98	2,515
Aluminum - 7049 Forging	-	-	-
Aluminum - 7075 Forging	304	1,216	2,986
Aluminum - 2024, 7075 Sheet, Plate, Extrusion	576	708	1,161
Steel	-	-	-
Titanium	-	-	-
Aluminum Honeycomb	79	95	776
Boron Aluminum	-	-	-
Other (Filler, Attachments, Paint, Balance Weights)	98	98	477
Subtotal	1,246	2,397	8,427
Advanced Composite			
Thornel 300 Graphite Epoxy			
12-Inch Tape	662	927	36,051
Broadgoods	107	150	4,167
Slit Tape	127	178	6,130
Cross-plied Tape	386	540	17,998
Subtotal	1,282	1,795	64,346
Total	2,528	4,192	72,773

[1]Cumulative average estimate of 300 quantity

TABLE 37
RESIZED ADVANCED COMPOSITE AIRCRAFT
RAW MATERIAL ESTIMATE – VERTICAL STABILIZER[1]

Material Category	Material Weight-Lb		Cost January 1973 Dollars
	Design	Purchased	
Conventional			
Glass & Fiberglass	118	201	559
Adhesive	99	119	3,054
Aluminum - 7049 Forging	-	-	-
Aluminum - 7075 Forging	401	1,604	3,946
Aluminum - 2024, 7075 Sheet, Plate, Extrusion	648	797	1,307
Steel	91	257	725
Titanium	-	-	-
Aluminum Honeycomb	133	160	1,307
Boron Aluminum	-	-	-
Other (Filler, Attachments, Paint, Balance Weights)	39	39	199
Subtotal	1,529	3,177	11,097
Advanced Composite			
Thornel 300 Graphite Epoxy			
12-Inch Tape	848	1,187	46,162
Broadgoods	-	-	-
Slit Tape	20	28	964
Cross-plied Tape	347	486	16,199
Subtotal	1,215	1,701	63,325
Total	2,744	4,878	74,422

[1]Cumulative average estimate of 300 quantity

TABLE 38
RESIZED ADVANCED COMPOSITE AIRCRAFT
RAW MATERIAL ESTIMATE – FUSELAGE[1]

Material Category	Material Weight-Lb		Cost January 1973 Dollars
	Design	Purchased	
Conventional			
Glass & Fiberglass	2,140	3,638	10,114
Adhesive	655	786	20,169
Aluminum - 7049 Forging	2,133	8,532	22,524
Aluminum - 7075 Forging	-	-	-
Aluminum - 2024, 7075 Sheet, Plate, Extrusion	1,696	2,284	3,746
Steel	608	1,632	4,602
Titanium	218	589	5,413
Aluminum Honeycomb	590	708	5,784
Boron Aluminum	2,134	3,201	24,711
Other (Filler, Attachments, Paint, Balance Weights)	1,323	1,323	6,444
Subtotal	11,497	22,864	103,507
Advanced Composite			
Thornel 300 Graphite Epoxy			
12-Inch Tape	2,248	3,148	122,426
Broadgoods	1,282	1,794	49,837
Slit Tape	5,557	7,780	267,943
Cross-plied Tape	1,369	1,919	63,961
Subtotal	10,456	14,641	504,167
Total	21,953	37,505	607,674

[1]Cumulative average estimate of 300 quanity

TABLE 39
RESIZED ADVANCED COMPOSITE AIRCRAFT
RAW MATERIAL ESTIMATE – WING[1]

MATERIAL CATEGORY	MATERIAL WEIGHT - LB		COST JANUARY 1973 DOLLARS
	DESIGN	PURCHASED	
CONVENTIONAL[2]			
SUBTOTAL	10,215	26,155	128,011
ADVANCED COMPOSITE			
Pitch Based Fibers Graphite Epoxy			
12 Inch Tape	2,272	3,181	17,686
Broadgoods	473	662	7,355
Slit Tape	566	792	9,678
Cross-plied Tape	1,385	1,939	32,323
SUBTOTAL	4,696	6,574	67,042
TOTAL	14,911	32,729	195,053

TABLE 40
RESIZED ADVANCED COMPOSITE AIRCRAFT
RAW MATERIAL ESTIMATE – HORIZONTAL STABILIZER[1]

MATERIAL CATEGORY	MATERIAL WEIGHT - LB		COST JANUARY 1973 DOLLARS
	DESIGN	PURCHASED	
CONVENTIONAL[2]			
SUBTOTAL	1,246	2,397	8,427
ADVANCED COMPOSITE			
Pitch Based Fibers Graphite Epoxy			
12 Inch Tape	662	927	5,154
Broadgoods	107	150	1,667
Slit Tape	127	178	2,175
Cross-plied Tape	386	540	9,002
SUBTOTAL	1,282	1,795	17,998
TOTAL	2,528	4,192	26,425

[1] Cumulative average estimate of 300 quantity

[2] Identical to conventional materials in the resized advanced composite aircraft with Thornel 300 Graphite Epoxy

TABLE 41
RESIZED ADVANCED COMPOSITE AIRCRAFT
RAW MATERIAL ESTIMATE – VERTICAL STABILIZER[1]

MATERIAL CATEGORY	MATERIAL WEIGHT - LB		COST JANUARY 1973 DOLLARS
	DESIGN	PURCHASED	
CONVENTIONAL[2]			
SUBTOTAL	1,529	3,177	11,097
ADVANCED COMPOSITE			
Pitch Based Fibers Graphite Epoxy			
12 Inch Tape	848	1,187	6,599
Broadgoods	-	-	-
Slit Tape	20	28	342
Cross-plied Tape	347	486	8,102
SUBTOTAL	1,215	1,701	15,043
TOTAL	2,744	4,878	26,140

TABLE 42
RESIZED ADVANCED COMPOSITE AIRCRAFT
RAW MATERIAL ESTIMATE – FUSELAGE[1]

MATERIAL CATEGORY	MATERIAL WEIGHT - LB		COST JANUARY 1973 DOLLARS
	DESIGN	PURCHASED	
CONVENTIONAL[2]			
SUBTOTAL	11,497	22,693	103,507
ADVANCED COMPOSITE			
Pitch Based Fibers Graphite Epoxy			
12 Inch Tape	2,248	3,148	17,503
Broadgoods	1,282	1,794	19,931
Slit Tape	5,557	7,780	95,072
Cross-plied Tape	1,369	1,919	31,989
SUBTOTAL	10,456	14,641	164,495
TOTAL	21,953	37,505	268,002

[1]Cumulative average estimate of 300 quantity

[2]Identical to conventional materials in the resized advanced composite aircraft with Thornel 300 Graphite Epoxy

TABLE 43
RESIZED ADVANCED COMPOSITE AIRCRAFT
RAW MATERIAL AND PURCHASED PARTS – SUMMARY[1]

AIRCRAFT COMPONENT	DESIGN COST WEIGHT - LB	COST JANUARY 1973 DOLLARS
WING		
Wing Box	6,249	237,569
Aileron and Balance Weight	237	5,351
Spoiler	307	13,498
Remainder	8,118	105,595
Subtotal	14,911	362,013
HORIZONTAL STABILIZER		
Horizontal Box	1,344	52,314
Elevator	508	17,311
Remainder	676	3,148
Subtotal	2,528	72,773
VERTICAL STABILIZER		
Vertical Box	1,080	46,661
Trailing Edge	145	3,864
Rudder	562	19,118
Remainder	957	4,779
Subtotal	2,744	74,422
FUSELAGE		
Primary Structure	11,254	351,708
Cargo Floor, Ramp, Supports	5,372	112,221
Remainder	5,327	143,745
Subtotal	21,953	607,674
REMAINDER OF AIRCRAFT[2]	30,959	478,626
TOTAL	73,095	1,595,508

[1]Cumulative average estimate of 300 quantity

[2]Includes the following airframe systems:

- landing gear (less rolling assembly)
- flight controls
- propulsion (less engine)
- fuel system
- auxiliary power unit
- instruments
- hydraulics
- pneumatic
- electrical
- avionics
- furnishing
- air conditioning
- ice protection
- handling gear

TABLE 44
RESIZED ADVANCED COMPOSITE AIRCRAFT WITH PITCH BASED FIBERS
RAW MATERIAL AND PURCHASED PARTS – SUMMARY[1]

AIRCRAFT COMPONENT	DESIGN COST WEIGHT - LB	COST JANUARY 1973 DOLLARS
WING		
Wing Box	6,249	83,775
Aileron and Balance Weight	237	1,951
Spoiler	307	3,732
Remainder	8,118	105,595
Subtotal	14,911	195,053
HORIZONTAL STABILIZER		
Horizontal Box	1,344	17,997
Elevator	508	5,280
Remainder	676	3,148
Subtotal	2,528	26,425
VERTICAL STABILIZER		
Vertical Box	1,080	13,493
Trailing Edge	145	1,998
Rudder	562	5,870
Remainder	957	4,779
Subtotal	2,744	26,140
FUSELAGE		
Primary Structure	11,254	146,301
Cargo Floor, Ramp, Supports	5,372	61,881
Remainder	5,327	59,820
Subtotal	21,953	268,002
REMAINDER OF AIRCRAFT[2]	30,959	478,626
TOTAL	73,095	994,246

[1]Cumulative average estimate of 300 quantity

[2]Includes the following airframe systems:

- landing gear (less rolling assembly)
- flight controls
- propulsion (less engine)
- fuel system
- auxiliary power unit
- instruments
- hydraulics
- pneumatic
- electrical
- avionics
- furnishing
- air conditioning
- ice protection
- handling gear

TABLE 45
DEVELOPMENT AND PRODUCTION COST ELEMENT
300 AIRCRAFT PROGRAM – JANUARY 197' DOLLARS

RESOURCE ELEMENT	BASELINE METAL AIRCRAFT	UNRESIZED ADVANCED COMPOSITE AIRCRAFT[1]	RESIZED ADVANCED COMPOSITE AIRCRAFT[1]	RESIZED ADVANCED COMPOSITE AIRCRAFT[2]	RESIZED ADVANCED COMPOSITE AIRCRAFT[1] WITH FIXED ENGINES
LABOR					
MANUFACTURING	$1344.573 M	$1140.337 M	$1093.564 M	$1093.564 M	$1071.197 M
TOOLING	269.510	226.880	217.658	217.638	215.707
PLANNING	142.258	123.782	118.096	118.096	116.867
QUALITY ASSURANCE	138.235	160.438	161.534	161.534	158.755
ENGINEERING DESIGN	298.984	318.065	318.065	318.065	318.065
ENGINEERING LABORATORY	50.082	59.567	59.567	59.567	59.567
FLIGHT TEST	38.829	38.829	38.829	38.829	38.829
PRODUCT SUPPORT	20.265	20.265	20.265	20.265	20.265
SUBTOTAL[3]	$2302.736 M	$2088.163 M	$2027.579 M	$2027.579 M	$1999.252 M
MATERIAL					
MANUFACTURING - RAW MATERIAL & PURCHASED PARTS	$ 257.254 M	$ 551.218 M	$ 527.698 M	$ 328.837 M	$ 525.276 M
EQUIPMENT - INSTRUMENTS AND SPECIAL EQUIPMENT	320.784	320.784	320.784	320.784	320.784
TOOLING	16.516	13.855	13.272	13.272	13.266
PRODUCT SUPPORT	20.468	20.468	20.468	20.468	20.468
FLIGHT TEST	5.266	5.266	5.266	5.266	5.266
SUBTOTAL[4]	$ 620.288 M	$ 911.591 M	$ 887.488 M	$ 688.627 M	$ 885.060 M
SUBCONTRACTS					
ENGINES	$ 450.000 M	$ 450.000 M	$ 415.607 M	$ 415.607 M	$ 450.000 M
AVIONICS	134.100	134.100	134.100	134.100	134.100
SUBTOTAL	$ 584.100 M	$ 584.100 M	$ 549.707 M	$ 549.707 M	$ 584.100 M
TOTAL PRICE	$3507.124 M	$3583.854 M	$3464.774 M	$3265.913 M	$3468.412 M

[1] ADVANCED COMPOSITE MATERIAL - THORNEL 300
[2] ADVANCED COMPOSITE MATERIAL - PITCH BASED FIBERS
[3] INCLUDES OVERHEAD, G&A, OVERTIME PREMIUM, DIRECT CHARGE, PROFIT
[4] INCLUDES DIRECT CHARGE AND PROFIT

The nonstructural portions of the airplane were held at a constant $15.46 per pound while the raw material and purchased parts going into the four major structural components varied between a low of $5.61 per pound of finished baseline structure to a high of $26.51 per pound of finished Thornel 300 structure. When the constant cost per pound of the nonstructural items was included, the variation in total raw materials and purchased parts dropped from a factor of approximately 6 to a factor of approximately 2-1/2. Material costs included a factor for Douglas internal distribution, warehousing, and handling.

6.3.4 Air Vehicle Costs

The comparative air vehicle costs of the five alternative configurations are shown in Table 45. The costs of instruments, special equipment, engines, and avionics were held constant for all configurations, excepting the scaled engine prices for the completely resized aircraft. Table 45 shows the major economy in manufacturing and, to a lesser extent, in tooling. Cost increases are indicated in quality assurance, laboratory, engineering, and material areas. Comments regarding laboratory and engineering increases have been given. Quality assurance increases stem from the same material complexity and from the fact that composite material physical properties are created each time a component is fabricated. Inspection techniques required are more sophisticated.

Several cost items were held constant over the spectrum of configurations. Flight test was not varied because the aircraft are not significantly different in size. Neither flight test hours nor instrumentation were changed. Product support labor was not changed. Similarly, product support material delivered with the aircraft (manuals, etc.) for maintenance and flight crews was considered constant over the range of configurations. The costs shown in Table 45 include overhead, general and administrative expense, overtime premium, direct charges, and profit, as applicable.

6.3.5 Other Acquisition Costs

A number of support items must also be acquired in addition to the air vehicle. Program management (embracing configuration control, schedule control, and all the other associated direct activities) is required during the development and production phases. Program management was estimated proportional to air vehicle costs based upon detailed estimates for other Air Force programs. A detailed estimate was made for test spares. Other estimates were proportional to the air vehicle costs. Product support, training and training equipment, and aerospace ground equipment were treated as constant amounts during both research and development and the production phases.

A detailed estimate was made for initial spares for the baseline configuration production phase. This estimate considered the number of squadrons, bases, and operational aircraft. Separate initial spares factors were developed for the air vehicle, avionics, and engines. The same factors were used to calculate spares costs for all five configurations. Packaging, marking, and shipping were taken proportional to spares costs. The cost of these activities increased the air vehicle costs by about 17 percent to provide an estimate of the total acquisition cost for deploying the 16-squadron force.

TABLE 46

MAINTENANCE MAN-HOURS PER FLIGHT HOUR

MAINTENANCE FUNCTIONS	BASELINE METAL AIRCRAFT	UNRESIZED ADVANCED COMPOSITE AIRCRAFT[1]	RESIZED ADVANCED COMPOSITE AIRCRAFT[1]	RESIZED ADVANCED COMPOSITE AIRCRAFT[2]	RESIZED ADVANCED COMPOSITE AIRCRAFT WITH FIXED ENGINES[1]
AIRFRAME	3.13	4.34	4.27	4.27	4.30
PROPULSION	3.62	3.62	3.34	3.34	3.62
AVIONICS	1.77	1.77	1.77	1.77	1.77
SUBTOTAL	8.52	9.73	9.38	9.38	9.69
SERVICING	2.70	2.70	2.70	2.70	2.70
CLEANING/ CORROSION CONTROL	0.28	0.28	0.28	0.28	0.28
SUPPORT OTHER	0.45	0.45	0.45	0.45	0.45
SUBTOTAL	3.43	3.43	3.43	3.43	3.43
PRE/POST FLIGHT	0.57	0.57	0.57	0.57	0.57
PHASE (PH) INSPECTION (LOOK)	0.98	0.98	0.98	0.98	0.98
SUBTOTAL	1.55	1.55	1.55	1.55	1.55
TOTAL	13.50	14.71	14.36	14.36	14.67

[1] ADVANCED COMPOSITE MATERIAL - THORNEL 300

[2] ADVANCED COMPOSITE MATERIAL - PITCH-BASED FIBERS

These results will be shown combined with operating and maintenance costs to obtain life-cycle costs.

6.4 LIFE-CYCLE COSTS

6.4.1 Operating Factors and Maintenance Manpower

The operational system costs were projected using the Air Force Planning Aircraft Cost Estimating (PACE) model for a force of 256 aircraft operating for 20 full-force years. Out of a total of 300 aircraft, 44 aircraft were withheld for pipeline, advanced attrition, and command and support purposes. The remaining 256 unit equipment (UE) aircraft are organized into 16 squadrons of 16 aircraft each. Each UE aircraft of the squadron was assumed to operate 900 hours per year.

The PACE operating and support cost model required estimates of maintenance man-hours per flying hour including both on- and off-airplane maintenance. These estimates were developed using standard maintenance engineering estimating techniques. The maintenance man-hours per flying hour (Table 46) varied from a low of 13.5 to a high of 14.7, a 6 to 9 percent increase across the configurations. The airframe maintenance for the composite aircraft reflects an increase of 36 to 38 percent in man-hours due to the present low confidence in the performance and reliability of large fuselage and wing composite structures. These additional man-hours will be required to accomplish repairs due to possible bond deterioration, erosion, and ground damage, which may be more time consuming and difficult to repair because of restricted accessibility to internally damaged areas. Repairs may also require additional skills, techniques, adhesive systems, and improved inspection techniques. This estimate is preliminary and represents the best current estimate tempered by a conservative approach with the new materials.

Propulsion maintenance man-hours per flying hour were estimated as a function of engine thrust. Avionics system maintenance was held constant. The servicing and inspection maintenance man-hours per flying hour were also held constant across all five configurations. For the total aircraft, an increase of approximately 8/10 maintenance man-hour per flying hour was estimated for the resized composite aircraft (Table 46, 13.50 to 14.36). The 36- to 38-percent increase in airframe maintenance man-hours per flying hour projected for advanced composite material was diluted by the effect of the constant maintenance items, reducing the percentage increase. Tables 47 and 48 display the total maintenance costs for all five configurations. Table 47 was prepared using the maintenance man-hours per flying hour factors originally estimated. Table 48 was prepared using equal base maintenance man-hours per flying hour. There is sufficient evidence from Air Force field maintenance experience with advanced composites to show that organization and field maintenance labor for secondary structures is comparable to conventional metals. If this trend is applicable to primary structures, the life-cycle costs shown in Table A-2, Appendix A, will be appropriate.

TABLE 47
COMPARISON OF MAINTENANCE COST ELEMENTS
256 OPERATING AIRCRAFT – JANUARY 1973 DOLLARS

MAINTENANCE COST ELEMENT	BASELINE METAL AIRCRAFT	UNRESIZED ADVANCED COMPOSITE AIRCRAFT[1]	RESIZED ADVANCED COMPOSITE AIRCRAFT[1]	RESIZED ADVANCED COMPOSITE AIRCRAFT[2]	RESIZED ADVANCED COMPOSITE AIRCRAFT WITH FIXED ENGINES[1]
REPLENISHMENT SPARES	290.3	290.3	281.0	267.3	285.7
MODIFICATION/SPARES	233.9	239.6	230.9	216.4	231.3
COMMON AGE/SPARES	31.7	31.7	31.7	31.7	31.7
SYSTEM SUPPORT MATERIAL	290.3	285.7	271.9	267.2	281.1
GENERAL SUPPORT MATERIAL	188.9	188.9	179.7	170.5	184.3
SUBTOTAL	1,035.1	1,036.2	995.2	953.1	1,014.1
MAINTENANCE PERSONNEL	588.1	610.9	598.2	598.2	610.9
DEPOT MAINTENANCE	753.9	752.5	730.0	730.0	753.5
SUBTOTAL	1,342.0	1,363.4	1,328.2	1,328.2	1,364.4
TOTAL	2,377.1	2,399.6	2,323.4	2,281.3	2,378.5
COMPARISON WITH BASELINE	1.000	1.009	0.977	0.960	1.001

[1]ADVANCED COMPOSITE MATERIAL - THORNEL 300

[2]ADVANCED COMPOSITE MATERIAL - PITCH BASED FIBERS

TABLE 48
COMPARISON OF MAINTENANCE COST ELEMENTS
EQUIVALENT MAINTENANCE MANPOWER TO BASELINE
256 OPERATING AIRCRAFT — JANUARY 1973 DOLLARS

MAINTENANCE COST ELEMENT	BASELINE METAL AIRCRAFT	UNRESIZED ADVANCED COMPOSITE AIRCRAFT[1]	RESIZED ADVANCED COMPOSITE AIRCRAFT[1]	RESIZED ADVANCED COMPOSITE AIRCRAFT[2]	RESIZED ADVANCED COMPOSITE AIRCRAFT WITH FIXED ENGINES[1]
REPLENISHMENT SPARES	290.3	290.3	281.0	267.3	285.7
MODIFICATION/SPARES	233.9	239.6	230.9	216.4	231.3
COMMON AGE/SPARES	31.7	31.7	31.7	31.7	31.7
SYSTEM SUPPORT MATERIAL	290.3	285.7	271.9	267.2	281.1
GENERAL SUPPORT MATERIAL	188.9	188.9	179.7	170.5	184.3
SUBTOTAL	1,035.1	1,036.2	995.2	953.1	1,014.1
MAINTENANCE PERSONNEL	588.1	588.1	588.1	588.1	588.1
DEPOT MAINTENANCE	753.9	752.5	730.0	730.0	753.5
SUBTOTAL	1,342.0	1,340.6	1,318.1	1,318.1	1,341.6
TOTAL	2,377.1	2,376.8	2,313.3	2,271.2	2,355.7
COMPARISON WITH BASELINE	1.000	1.000	0.973	0.955	0.991

[1]ADVANCED COMPOSITE MATERIAL - THORNEL 300

[2]ADVANCED COMPOSITE MATERIAL - PITCH BASED FIBERS

6.4.2 Total Life-Cycle Costs

The total life-cycle costs for the five configurations are shown in Table 49. From the basic maintenance man-hours per flying hour, maintenance and total personnel complements were developed on a squadron basis.

Materials and spares costs were based on the flyaway cost of each configuration, which takes into account the degree of application of advanced composite materials. Depot maintenance costs were based on both the MMH/FH and the material costs.

Petroleum, oil, and lubricants were calculated on the basis of a standard mission for all five configurations. The cost per gallon for jet fuel in fiscal year 1973 was 15 cents per gallon as reported by AFM 173-10. This figure, consistent with the other economic factors, was used despite the fact that current fuel prices are significantly higher. With these exceptions, the PACE model was used in a constant configuration to produce the 20-year operating and support costs shown in Table 49. Appendix A includes the sensitivity of life-cycle cost to increased fuel costs and alternate maintenance concepts.

6.5 COST TRENDS AND CONCLUSIONS

The configuration characteristics and the development, procurement, operations, and support costs for the five configurations are summarized in Table 50. Although significant weight savings (up to 40 percent) are indicated for some components, the avionics, engines, systems, and nonstructural portions of the aircraft weight dilute the impact of the structural weight saving. The aircraft manufacturer's planning report (AMPR) weight of the resized aircraft is 12 percent less than the baseline. The operating weight empty is only 10.4 percent less and takeoff gross weight (TOGW) is only 7.7 percent less than the baseline aircraft.

A reverse trend is indicated in the cost analysis. Although the material costs of the advanced composite aircraft are considerably higher than the baseline metals, the structural cost impact is diluted by the effect of the avionics, engines, aircraft systems, and other nonstructural portions of the aircraft. Thus the total unit price of the aircraft (even with the more expensive Thornel 300 material) may be the same or lower than the unit price of the baseline metal airplane (see Table 51). The lower priced pitch-based composite material yields an aircraft price that is 7.5 percent less than the cost of the baseline metal airplane.

6.5.1 Cost Trends

The cost analysis results indicate several important points:

- A comparison of the costs of the baseline metal configuration and the unresized composite configuration suggests that total airplane price and cost will increase unless resizing benefits are taken.
- A comparison of the resized configurations shows that reduced cost fibers are necessary for reduced unit and system costs.

TABLE 4?

LIFE CYCLE COST COMPARISON

300 AIRCRAFT PROGRAM – JANUARY 1973 DOLLARS

RESOURCE ELEMENT	BASELINE METAL AIRCRAFT	UNRESIZED ADVANCED COMPOSITE AIRCRAFT[1,3]	RESIZED ADVANCED COMPOSITE AIRCRAFT[1]	RESIZED ADVANCED COMPOSITE AIRCRAFT[2]	RESIZED ADVANCED COMPOSITE AIRCRAFT[1] WITH FIXED ENGINES
DEVELOPMENT					
AIR VEHICLE	481.8	485.1	477.3	463.8	475.5
PROJECT MANAGEMENT	31.6	31.8	31.3	30.4	31.2
PRODUCT SUPPORT	26.8	26.9	26.9	26.9	26.9
TEST SPARES	30.8	31.1	30.5	29.7	30.5
PKG. MRKG. SHPG.	.9	.9	.9	.9	.9
ECPS	19.3	19.4	19.1	18.6	19.0
TRAINING/TRAINERS	27.5	28.1	27.1	25.3	27.2
AGE	40.3	41.2	39.8	37.3	39.8
SUBTOTAL	659.0	664.5	652.9	632.9	651.0
PRODUCTION					
AIR VEHICLE (PME)	2985.0	3058.1	2946.7	2761.4	2952.2
PROJECT MANAGEMENT	52.2	53.5	51.6	48.3	51.7
PRODUCT SUPPORT	13.8	13.9	13.9	13.9	13.9
INITIAL SPARES	294.8	299.2	286.1	275.0	292.9
PKG. MRKG. SHPG.	8.9	9.0	8.6	8.2	8.8
ECP	119.4	122.3	117.9	110.6	118.1
TRAINING/TRAINERS	41.2	42.2	40.6	38.1	40.7
AGE	60.5	61.9	59.6	55.9	59.7
SUBTOTAL	3575.8	3660.1	3525.0	3311.2	3538.0
ACQUISITION TOTAL	4234.8	4324.6	4177.9	3944.1	4189.0
OPERATING & SUPPORT (20 YRS)					
DIRECT					
MATERIALS/SPARES	1035.1	1036.2	995.3	953.1	1014.1
PERSONNEL	1309.8	1332.6	1319.9	1319.9	1332.6
POL	1566.7	1516.0	1465.3	1465.4	1520.6
DEPOT MAINTENANCE	753.9	752.5	730.0	730.0	753.5
MISCELLANEOUS	15.0	15.4	15.2	15.2	15.4
INDIRECT					
BASE OPERATING SUPPORT	645.5	665.9	658.4	658.3	665.9
PLANNING ADDITIVES	98.5	100.8	99.6	99.6	100.8
SUBTOTAL	5424.5	5419.4	5283.7	5241.6	5402.9
LIFE CYCLE COST	9659.3	9744.0	9461.6	9185.7	9591.9

[1]ADVANCED COMPOSITE MATERIAL - THORNEL 300
[2]ADVANCED COMPOSITE MATERIAL - PITCH BASED FIBERS
[3]THIS AIRCRAFT EXCEEDS BASELINE MISSION PERFORMANCE.

TABLE 50
CONFIGURATION CHARACTERISTICS AND COST SUMMARY

CHARACTERISTIC	CONFIGURATION				
	BASELINE METAL AIRCRAFT	UNRESIZED ADVANCED COMPOSITE AIRCRAFT[1]	RESIZED ADVANCED COMPOSITE AIRCRAFT[1]	RESIZED ADVANCED COMPOSITE AIRCRAFT[2]	RESIZED ADVANCED COMPOSITE AIRCRAFT WITH FIXED ENGINES[1]
THRUST/ENGINE - SLS, LB	14,900	14,900	13,760	13,760	14,900
WEIGHT SUMMARY - LB					
AMPR WEIGHT	79,016	73,269	70,064	70,064	70,033
MFG. WEIGHT EMPTY	98,724	92,977	88,487	88,487	89,515
OPERATORS WEIGHT EMPTY	103,234	97,487	92,980	92,980	94,000
TAKEOFF GROSS WEIGHT	150,000	150,000	138,500	138,500	139,890
COST WEIGHT	82,055	76,309	73,095	73,095	73,072
COST SUMMARY - JAN. 1, 1973 DOLLARS					
RDT&E	659.0	664.5	652.9	632.9	651.0
PRODUCTION	3575.8	3660.1	3525.0	3311.2	3538.0
ACQUISITION SUBTOTAL	4234.8	4324.6	4177.9	3944.1	4189.0
OPERATIONS & SUPPORT (20 YRS)	5424.5	5419.4	5283.7	5241.6	5402.9
TOTAL LIFE CYCLE	9659.3	9744.0	9461.6	9185.7	9591.9
PRODUCTION UNIT PRICE	10.165	10.413	10.036	9.408	10.054

[1]ADVANCED COMPOSITE MATERIAL - THORNEL 300

[2]ADVANCED COMPOSITE MATERIAL - PITCH BASED FIBERS

TABLE 51

COST COMPARISON OF THE BASELINE METAL TO THE FOUR ADVANCED COMPOSITE AIRCRAFT

RESOURCE ELEMENT	BASELINE METAL AIRCRAFT	UNRESIZED ADVANCED COMPOSITE AIRCRAFT[1]	RESIZED ADVANCED COMPOSITE AIRCRAFT[1]	RESIZED ADVANCED COMPOSITE AIRCRAFT[2]	RESIZED ADVANCED COMPOSITE AIRCRAFT[1] WITH FIXED ENGINES
LABOR					
MANUFACTURING	1.000	0.848	0.813	0.813	0.797
TOOLING	1.000	0.842	0.808	0.808	0.800
PLANNING	1.000	0.870	0.830	0.840	0.822
QUALITY ASSURANCE	1.000	1.161	1.169	1.169	1.148
ENGINEERING DESIGN	1.000	1.064	1.064	1.064	1.064
ENGINEERING LABORATORY	1.000	1.189	1.189	1.189	1.189
FLIGHT TEST	1.000	1.000	1.000	1.000	1.000
PRODUCT SUPPORT	1.000	1.000	1.000	1.000	1.000
SUBTOTAL	1.000	0.907	0.881	0.881	0.868
MATERIAL					
MANUFACTURING - RAW MATERIAL & PURCHASED PARTS	1.000	2.143	2.051	1.278	2.042
EQUIPMENT - INSTRUMENTS AND SPECIAL EQUIPMENT	1.000	1.000	1.000	1.000	1.000
TOOLING	1.000	0.839	0.804	0.804	0.804
PRODUCT SUPPORT	1.000	1.000	1.000	1.000	1.000
FLIGHT TEST	1.000	1.000	1.000	1.000	1.000
SUBTOTAL	1.000	1.470	1.431	1.110	1.427
SUBCONTRACTS					
ENGINES	1.000	1.000	0.924	0.924	1.000
AVIONICS	1.000	1.000	1.000	1.000	1.000
SUBTOTAL	1.000	1.000	0.941	0.941	1.000
UNIT PRICE	1.000	1.020	0.987	0.925	0.989

[1] ADVANCED COMPOSITE MATERIAL - THORNEL 300 GRAPHITE EPOXY

[2] ADVANCED COMPOSITE MATERIAL - PITCH BASED FIBERS GRAPHITE EPOXY

- The favorable cost and weight trend data should be confirmed by design and manufacture of primary structural components and phasing into production on an orderly basis. The results indicate maximal application of composites is most promising for the AMST wing followed in order by the horizontal tail, vertical tail, and the fuselage.

- Maintenance cost data should be gathered from field users of composites, particularly for thicker solid laminate constructions as well as for sandwich constructions to provide a base for more definitive maintainability estimates.

The extensive use of advanced composite materials impacts the elements of cost quite differently. The use of composite materials leads to a completely new component design concept reducing the number of parts and, therefore, revising the traditional mix of fabrication and assembly activities. A significant impact is on tooling costs. Because of the lower composite parts count, fewer tools must be built. The most significant impact is on manufacturing and planning labor, for which the recurring cost benefits can be great. Although fewer parts will be manufactured, more labor may be required per part because each one will be larger in size and contain integrated details normally separate parts.

The use of composite materials may have an adverse impact on the costs of quality assurance, engineering design, and engineering laboratories. The process of designing, testing, and qualifying a composite part is more complex than for a metal part. Each layer of the material must be carefully analyzed to determine both the form of the material and the orientation. The cost of raw materials and purchased parts will increase significantly because the costs of the composite materials, even at projected prices, will still be higher than the cost of current state-of-the-art metals.

6.5.2 Economic Benefits

Before the economic benefits of advanced composite technology can be realized, the composite materials must be used to a greater extent than they have been to date. In particular, advanced composite must be used in the primary structure to permit resizing of the entire vehicle. Unless the airplane is resized to take advantage of the composite material properties, a total system cost increase results (see Table 49). The airframe must be resized and the engine should be also. Out of a total cost of $9.66 billion for 20 years of operation, approximately $197 million can be saved by using the Thornel 300 material at the present 30- to 50-percent composite utilization. The shift from Thornel 300 material to the less expensive pitch-based fiber metal (or alternatively an equivalent drop in Thornel 300 prices) would produce additional savings on the order of $276 million for a total savings of $473 million over the life cycle of the system.

Extensive use of composite materials in future airframes will revise the traditional proportions between fabrication and assembly. The nature of the materials and the impact on the manufacturing process will have beneficial effects on tooling, manufacturing, and planning, in that order. The extensive use of composite material presently appears to have adverse cost effects on material, quality assurance, engineering design, and laboratory test. Unless the cost of design and manufacture of equipment items and engines (Table 45) can be significantly reduced, these items will reduce the potential economic impact of the composite structures.

Estimates of the economic benefits were made for a military AMST program of 300 airplanes. Two hundred and fifty-six of these aircraft were assumed deployed in 16 squadrons operating at 900 hours per year. To evaluate the results, a 10-percent discount rate was applied to roughly reflect the current decision criteria of defense procurement officials. The results are displayed in Table 52.

The resized pitch-based composite aircraft shows a present value savings of $205 million (Table 52). The program would break even if $205 million were expended for facilities and machine tools to develop and produce composite airplanes. The analysis to determine whether this amount would be sufficient to purchase the requisite facilities and machine tools has not been conducted. A complete manufacturing plan would be required to determine quantities and sizes of tools, fabrication equipment, autoclaves, etc., required to manufacture the composite AMST at the indicated production rates.

It is tempting to extrapolate from the presently reported economic results since the cost analysis is in constant January 1973 dollars. The current escalation of material and labor could have a further impact on reported results; however, the inflationary problem with regards to current escalations is too fluid to pin down labor and material price changes to gauge their effects. It may be offered that conventional materials are rising markedly and they are classed as being more energy intensive than the composites. We could therefore expect sizable increases in costs with the conventional materials but moderate increases with the composites, since the latter are showing a downward cost trend due to increased volume. It should also be considered that the conventional materials are going into short or limited supply as resources.

6.5.3 Diminishing Returns

The amount of advanced material that should be used in an airplane varies with the development status of the material. As a new material first appears, it is used only in the most effective applications. Gradually the price of the material drops, the technology improves, and it is used in greater quantities (up to some limit). This historical process recognizes that there is a constantly changing and increasing optimum usage of a given material for a given design. There is some point at which it ceases to be economic to substitute a stronger but more expensive material for a weaker but cheaper material. During this analysis an attempt was made to try to establish this point (the point of diminishing returns) for composite materials.

Table 53 displays the basic data used in this analysis of diminishing returns. The weight data show weight savings between 10 and 22 percent for the major components and an average of 15 percent for the airframe as a whole. The amount of composites used in each of these components varied between 32 and 5[illegible] percent. The costs represent the component at the completion of subassembly, including material, fabrication, subassembly, quality assurance, planning, and sustaining tooling. The component costs at this point varied from 90 to 109 percent of the baseline for the Thornel 300 material, and

TABLE 52

PRESENT VALUE COMPARISONS OF LIFE CYCLE COSTS (DOLLARS, MILLIONS)

CONFIGURATION	10% DISCOUNT RATE	
	LIFE CYCLE COST	DELTA LIFE CYCLE COST
BASELINE	3559.6	-
UNRESIZED ADVANCED COMPOSITE	3611.1	+51.5
RESIZED ADVANCED COMPOSITE WITH THORNEL 300	3500.4	-59.2
RESIZED ADVANCED COMPOSITE WITH PITCH BASED FIBER	3354.7	-204.9
RESIZED ADVANCED COMPOSITE WITH FIXED ENGINE	3527.4	-32.3

TABLE 53
ANALYSIS OF WEIGHT AND COST SAVINGS

	WEIGHT DATA - LB			COST DATA			Δ$/ΔLBS	
	BASELINE	COMPOSITE	SAVINGS PERCENT	BASELINE	RESIZED COMPOSITE THORNEL 300	RESIZED COMPOSITE PITCH BASE FIBER	RESIZED COMPOSITE THORNEL 300	RESIZED COMPOSITE PITCH BASE FIBER
WING - TOTAL CONVENTIONAL MATERIAL ADVANCED MATERIAL % ADVANCED MATERIAL	18,765 18,765 - 0.0	14,911 10,215 4,696 31.5	20.5	$2.094 M	$1.896 M 90.5% OF BASELINE COST	$1.729 M 82.6% OF BASELINE COST	-51.38	-94.71
HORIZONTAL STABILIZER - TOTAL CONVENTIONAL MATERIAL ADVANCED MATERIAL % ADVANCED MATERIAL	3,234 3,234 - 0.0	2,528 1,246 1,282 50.7	21.8	$0.301	$0.328 M 109.0% OF BASELINE COST	$0.282 M 93.7% OF BASELINE COST	+38.24	-26.91
VERTICAL STABILIZER - TOTAL CONVENTIONAL MATERIAL ADVANCED MATERIAL % ADVANCED MATERIAL	3,460 3,460 - 0.0	2,744 1,529 1,215 44.3	20.7	$0.305	$0.326 M 106.9% OF BASELINE COST	$0.277 M 90.8% OF BASELINE COST	+29.33	-39.11
FUSELAGE - TOTAL CONVENTIONAL MATERIAL ADVANCED MATERIAL % ADVANCED MATERIAL	24,367 24,367 - 0.0	21,953 11,497 10,456 47.6	9.9	$1.748	$1.885 M 107.8% OF BASELINE COST	$1.545 M 88.3% OF BASELINE COST	+56.75	-84.09
STRUCTURE - TOTAL CONVENTIONAL MATERIAL ADVANCED MATERIAL % ADVANCED MATERIAL	49,826 49,826 - 0.0	42,136 24,487 17,649 41.9	15.4	$4.448	$4.435 M 99.7% OF BASELINE COST	$3.833 M 86.2% OF BASELINE COST	-1.70	-79.97

from 83 to 94 percent for the pitch-based fiber material. The changes in cost divided by the changes in weight were calculated from this information. All of the components were reduced in weight and all were reduced in cost when pitch-based fiber was used. Only the wing was reduced in cost when the Thornel 300 was used for all components.

These data are plotted in Figure 70. For the major components (but not the fuselage) there was an increasing percentage of weight saved as the percentage of composites increased. Examination of structural subcomponent weight and utilization data (which lies in the higher 57- to 73-percent utilization range) shows an increasing rate of weight saving at higher utilization than displayed in Figure 70. Figure 70 displays data for which isolated costs were determined. The cost saved per pound saved for the Thornel 300 and the pitch-based fiber declined with increasing percentage of composite materials, following the law of diminishing returns. With costs appropriate to the Thornel 300, the best usage of composite material would be somewhat less than the 30- to 50-percent range on the airplane basis. With pitch-based fibers, this percentage would move up within the range of the 30- to 50-percent composite material postulated in the present designs.

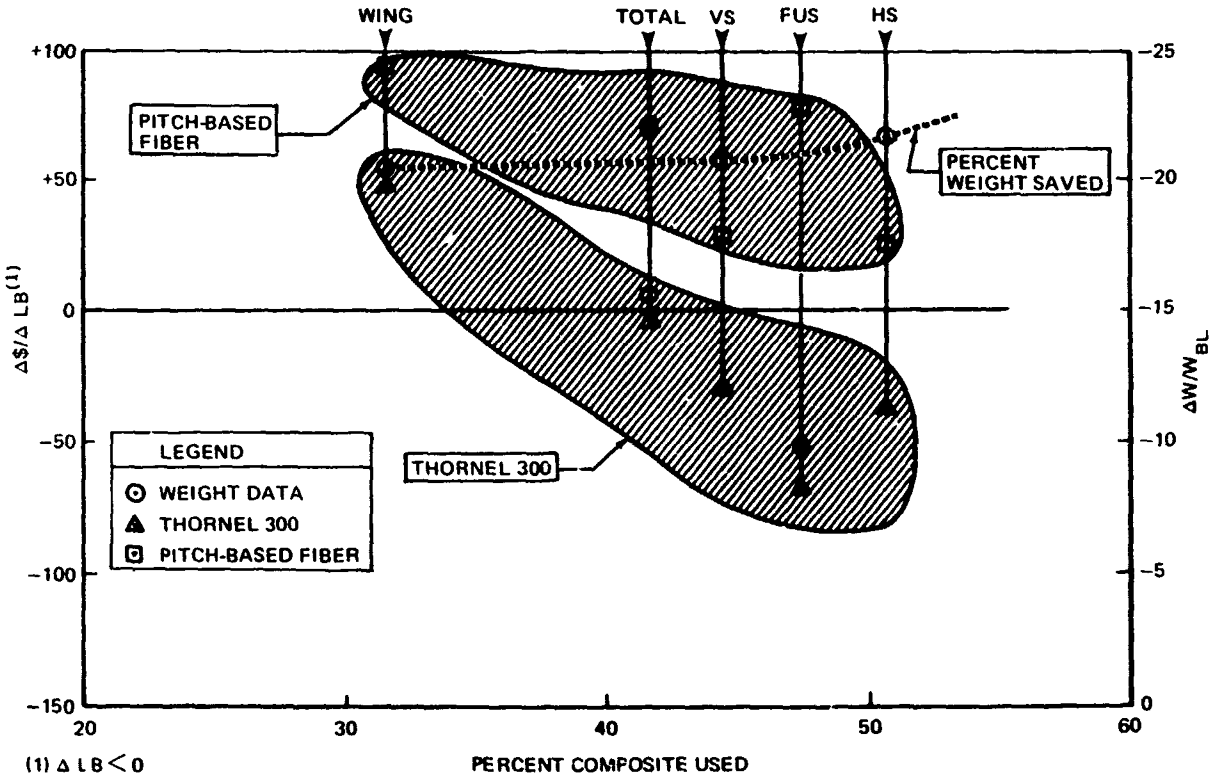


FIGURE 70. COST PER POUND SAVED AND PERCENT WEIGHT SAVED.

SECTION 7
REFERENCES

1. Nelson, W. D., et al.; "Composite Medium STOL Transport Structural Integration Study," Task VI of Composite Wing Conceptual Design, Report AFML-TR-72-228, October 1972.

2. Logan, T. R.; M. M. Platte; and R. L. Zwart; "A Study of the Costs and Benefits of the Application of Composite Materials to Civil STOL Aircraft," Final Report NASA CR-114701, December 1973.

3. Kulkarni, S. V.; J. S. Rice; B. W. Rosen; "An Investigation of the Compressive Strength of PRD-49-III/Epoxy Composites," Materials Sciences Corporation, Report NASA CR-112334, 1973.

4. I man, G. M. and A. Manno; "Development of a Graphite Horizontal Stabilizer," AIAA Paper 72-358, AIAA/ASME/SAE 13th Structures, Structural Dynamics, and Materials Conference, April 1972.

5. Nelson, W. D., et al., "Composite Wing Conceptual Design," Report AFML-TR-73-57, March 1973.

6. "Flight-Service Program for Advanced Composite Rudders on Transport Aircraft," Contract NASI-12954.

7. Douglas Aircraft Co. "AMST Design Analysis, Vol. I, Design to Cost Study Report," MDC J5363-5, January 1973.

8. Nelson, W. D., "Configurations and Joints for Conceptual Truss Web Wing Design," Douglas IRAD Technical Report MDC J6097 (in preparation).

9. AFML-TR-72-228, Composite Medium STOL Transport Structural Integration Study, October 1972.

10. Hitch, C. J. and R. N. McKean; "The Economics of Defense in the Nuclear Age," Howard University Press, Cambridge, Massachusetts, 1967.

11. Lehman, G. H., et al. "Study to Assess the Utility of Advanced Materials in Aircraft Structures" (U), Contract AF33(615)-5085, Douglas Aircraft Co., Report No. DAC 56087A (Secret), October 1967.

APPENDIX A

SENSITIVITY OF LIFE-CYCLE COST TO PETROLEUM, OIL, AND LUBRICANT (POL) COSTS AND MAINTENANCE ASSUMPTIONS

Although the operating costs in the body of this report were calculated by using jet fuel costs of 15 cents per gallon (1973), current jet fuel costs are ranging between 30 and 40 cents per gallon. While regulatory procedures govern domestic crude sources, the foreign supplies are not well defined as to price levels. To account for the uncertainties with this resource, Tables A-1 through A-4 have been prepared to exhibit the sensitivities of life-cycle cost to changes in fuel price over a range of 15 to 60 cents per gallon of jet fuel. This has been accomplished for four maintenance concepts or assumptions. Case 1 (Table A-1) shows the effect when baseline metal airplane and composite airplane maintenance materials and labor are each, respectively, the same as used to calculate life-cycle costs in the body of the report (Paragraph 6.4). Case 2 (Table A-2) assumes maintenance labor and material for the composite is the same as the baseline. Case 3 (Table A-3) uses composite airplane personnel costs which are the same as the baseline but composite airplane maintenance material costs are the same as the Case 1 composite airplane (best estimate). Case 4 (Table A-4) composite configurations are calculated using the Case 1 material costs (best estimates) and maintenance labor cost reduced in proportion to the average reduction of composite support material and depot costs.

Clearly the resized advanced composite airplane with pitch-based fiber material yields the lowest total life-cycle cost regardless of fuel price and maintenance concept. We can further conclude that as the price level increases, the savings in fuel cost with this airplane increases, as expected.

PRECEDING PAGE BLANK-NOT FILMED

TABLE A-1
SENSITIVITY OF LIFE CYCLE COST TO FUEL PRICES FOR A FIXED MAINTENANCE CONCEPT

CASE 1. MAINTENANCE CONCEPT SHOWN IN FINAL REPORT

AIRCRAFT CONFIGURATION	LIFE CYCLE COST FOR GIVEN FUEL PRICE LEVEL			
	$0.15/Gal	$0.30/Gal	$0.45/Gal	$0.60/Gal
Baseline Metal	$ 9.659B	$11.226B	$12.797B	$14.363B
Unresize Advanced Composite with Thornel 300	$ 9.744	$11.261	$12.778	$14.294
Resized Advanced Composite with Thornel 300	$ 9.462	$10.927	$12.392	$13.857
Resized Advanced Composite with Pitch Based Fibers	$ 9.186	$10.650	$12.116	$13.581
Resized Advanced Composite Thornell 300 and Fixed Engine	$ 9.594	$11.114	$12.635	$14.158

TABLE A-2
SENSITIVITY OF LIFE CYCLE COST TO FUEL PRICES FOR A FIXED MAINTENANCE CONCEPT

CASE 2. COMPOSITE CONFIGURATION MAINTENANCE AND PERSONNEL COSTS EQUIVALENT TO BASELINE

AIRCRAFT CONFIGURATION	LIFE CYCLE COST FOR GIVEN FUEL PRICE LEVEL			
	$0.15/Gal	$0.30/Gal	$0.45/Gal	$0.60/Gal
Baseline Metal	$ 9.659B	$11.226B	$12.797B	$14.363B
Unresized Advanced Composite with Thornel 300	$ 9.690	$11.220	$12.736	$14.252
Resized Advanced Composite with Thornel 300	$ 9.498	$10.963	$12.429	$13.894
Resized Advanced Composite with Pitch Based Fibers	$ 9.250	$10.715	$12.180	$13.646
Resized Advanced Composite Thornel 300 and Fixed Engines	$ 9.565	$11.086	$12.606	$14.127

TABLE A-3
SENSITIVITY OF LIFE CYCLE COST TO FUEL PRICES FOR A FIXED MAINTENANCE CONCEPT

CASE 3. COMPOSITE CONFIGURATION WITH BASELINE MAINTENANCE PERSONNEL COST AND MATERIAL SAME AS CASE 1.

AIRCRAFT CONFIGURATION	LIFE CYCLE COST FOR GIVEN PRICE LEVEL			
	\$0.15/Gal	\$0.30/Gal	\$0.45/Gal	\$0.60/Gal
Baseline Metal	\$ 9.659B	\$11.226B	\$12.797B	\$14.363B
Unresized Advanced Composite with Thornel 300	\$ 9.707	\$11.237	\$12.753	\$14.269
Resized Advanced Composite with Thornel 300	\$ 9.451	\$10.917	\$12.382	\$13.848
Resized Advanced Composite with Pitch Based Fibers	\$ 9.176	\$10.641	\$12.106	\$13.572
Resized Advanced Composite Thornell 300 and Fixed Engines	\$ 9.569	\$11.090	\$12.610	\$14.131

TABLE A-4
SENSITIVITY OF LIFE CYCLE COST TO FUEL PRICES FOR A FIXED MAINTENANCE CONCEPT

CASE 4. COMPOSITE CONFIGURATION WITH MAINTENANCE MATERIAL COST SAME AS CASE 1 AND MAINTENANCE PERSONNEL COST TREND SAME AS MATERIAL/DEPOT

AIRCRAFT CONFIGURATION	LIFE CYCLE COST FOR GIVEN PRICE LEVEL			
	\$0.15/Gal	\$0.30/Gal	\$0.45/Gal	\$0.60/Gal
Baseline Material	\$ 9.659B	\$11.226B	\$12.797B	\$14.363B
Unresized Advanced Composite with Thornel 300	\$ 9.720	\$11.237	\$12.754	\$14.270
Resized Advanced Composite with Thornel 300	\$ 9.431	\$10.896	\$12.361	\$13.826
Resized Advanced Composite with Pitch Based Fiber	\$ 9.143	\$10.607	\$12.073	\$13.538
Resized Advanced Composite Thornel 300 and Fixed Engines	\$ 9.565	\$11.085	\$12.606	\$14.129

APPENDIX B
PRELIMINARY DESIGN CONCEPT EVALUATION

Initial design activity evolved the design of a variety of composite structure elements. These were sketched so various engineering and production disciplines could evaluate the design concepts. The concepts included primarily wing box (and empennage box) geometry arrangements falling into four main categories: truss-web variations, truss-rib, truss-spar, and multirib designs. (Figures B-1, B-2, and B-3 are representative of each category.) Accompanying the box concepts was a series of cover stiffening and substructure stiffening concepts, Figures B-4 and B-5. The concepts ranged from plain honeycomb panels to various types of solid-laminated stiffened panels. Each panel type could be used in one or more of the four box designs.

The preliminary evaluation of box and panel stiffening concepts was aimed at choosing combinations that appeared to present the greatest potential for low-cost construction without going into detailed estimates. Long-term costs (maintainability and reliability) were also rated. Preliminary concept selection criteria were used as follows:

- Fabrication, assembly, and tooling costs
- Structural reliability (ease of incorporating or inherent, failsafe, and fatigue characteristics)
- Maintainability and repairability
- Environmental vulnerability
- Manufacturing feasibility
- Component weight
- Fuel volume (in the case of wing components)

Rating charts were prepared by the various groups to rate each design concept according to the above criteria. It is instructive for future evaluations of this type that there were major inconsistencies in the results.

B-1 PANEL RATINGS

Examination of ranking results revealed the following preferences for panel construction. Manufacturing, design, and analysis disciplines preferred the newer solid-laminate designs in large integral structures (fewer joints) from anticipated cost and reliability standpoints. Simple, constant thickness sandwich was also preferred from cost, analysis, and fabrication standpoints. However, the environmental vulnerability, fuel volume limitation, as well as the design complications of providing edge closures, hardpoints, doublers, and fasteners in sandwich was assessed against sandwich designs. The maintainability discipline, on the other hand, preferred the constant thickness

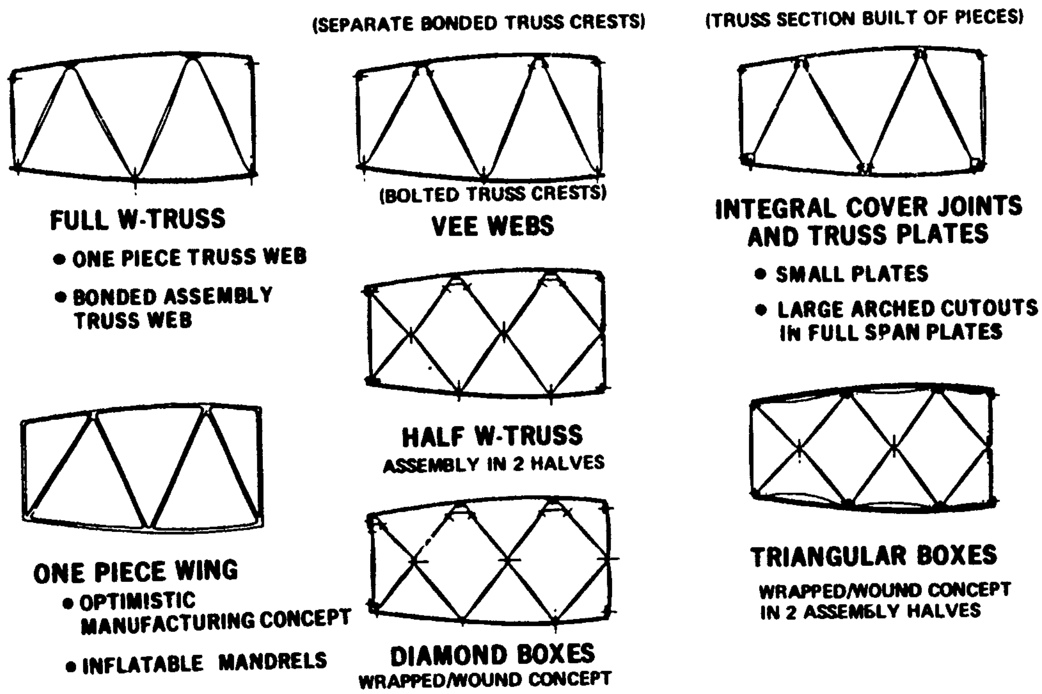


FIGURE B-1. TRUSS-WEB ALTERNATES

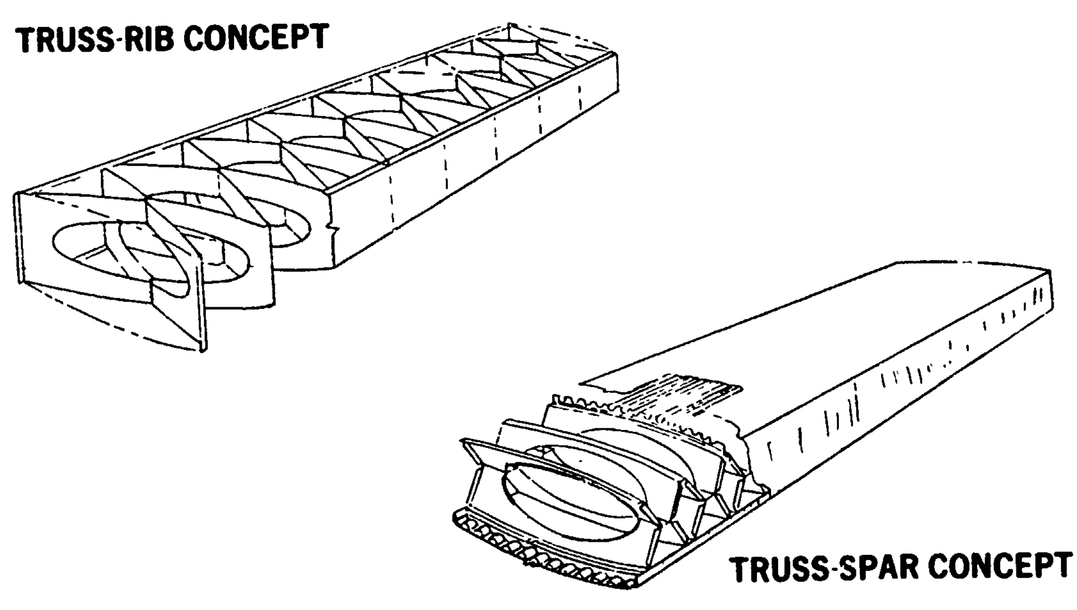
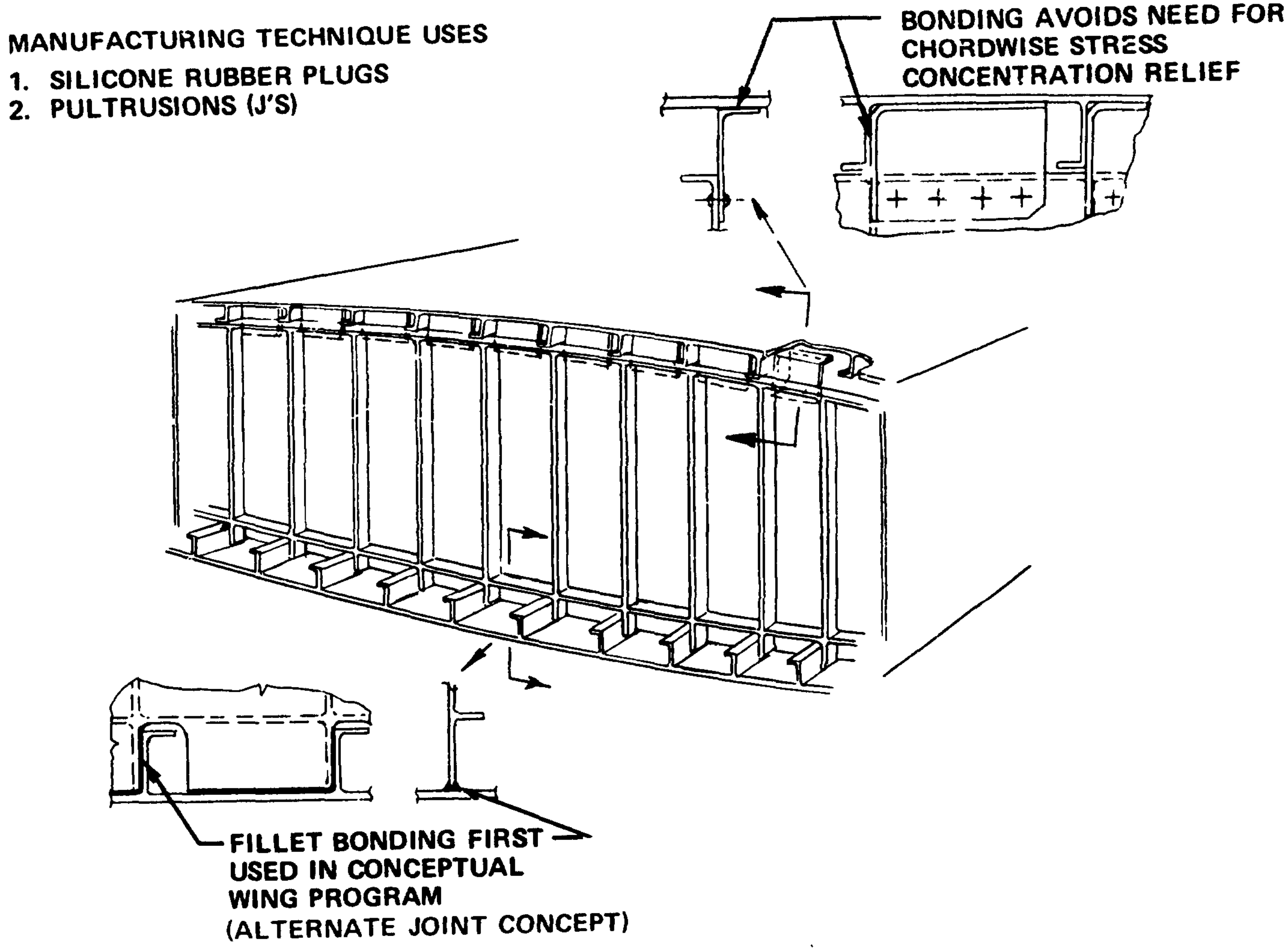


FIGURE B-2. ALTERNATE WING AND EMPENNAGE TRUSS CONCEPTS



FIBURE B-3. MULTIRIB SOLID LAMINATE WING CONCEPT

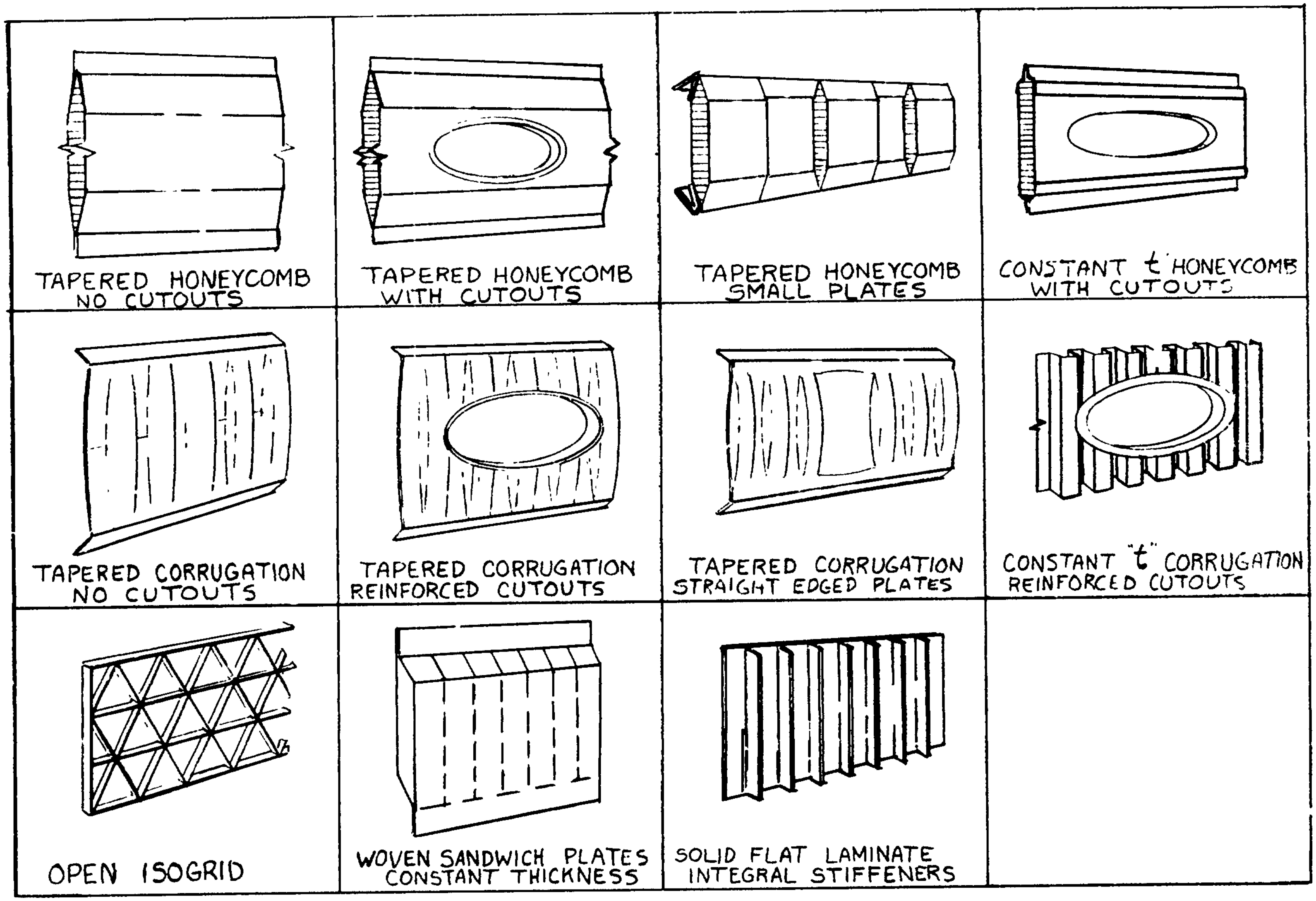


FIGURE B-4. SUBSTRUCTURE STIFFENING CONCEPTS

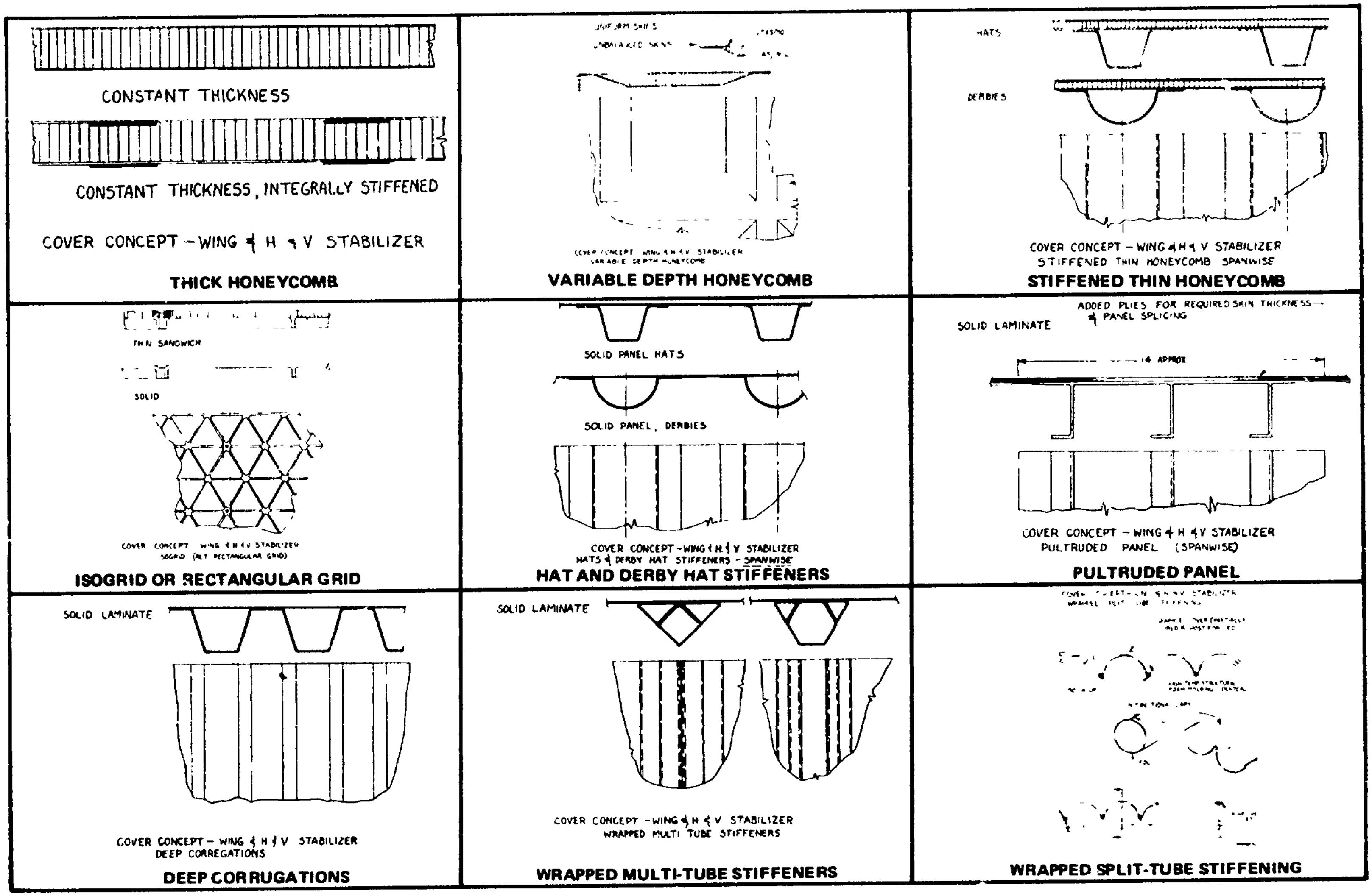


FIGURE B-5. COVER STIFFENING CONCEPTS

sandwich with cutouts and many separate assembly joints for ease of repair-or-replace situations. This preference stemmed partly from familiarity with sandwich construction and corresponding unfamiliarity with solid-laminate construction. Repair characteristics of solid laminate stiffened were not, on the other hand, considered more difficult than sandwich, merely that a positive rating could not be made without data.

Since long-term cost implications of vulnerability and maintainability were too ill-defined to use quantitatively, it was decided to base the design selections for both panels and box geometries on cost estimates of initial fabrication (near-term costs), rather than ambiguous results of complex rating charts. With adequate program time, long-term costs might be evaluated on a parametric basis for design selection.

B-2 TRUSS-WEB BOX CONCEPTS

The first sketch in Figure B-1 shows the full W-truss cross section of the truss-web wing concept developed in the Composite Wing Conceptual Design program, Reference 5. Key ideas of truss web are the V-joint and the monolithic (truss web) substructure which are particularly adapted to fibrous construction. Key difficulties with truss web in this thick wing application are:

- The fabrication cost aspect of the truss web with required lightening holes
- The avoidance of chordwise crossing members to introduce the concentrated loads from pylons and flaps

It was recognized that hand layup of tapes or broadgoods is the feasible layup procedure for the large W-truss and it is difficult to introduce automated layup concepts for that substructure geometry. The variations suggested in Figure B-1 were studied from the standpoint of reducing truss web fabrication cost without increasing box assembly cost. In all cases, "cutout" or lightened versions of the truss webs were assumed, since it was shown previously in Reference 5 and again in Reference 2 that the weight of the optimum uncut truss web in a deep wing produced a 7- to 8-percent higher weight than the equivalent optimum multirib configuration. The most feasible cost and weight configuration, that also could be adapted to chordwise load inputs, appeared to be the truss web with individual truss plates connecting upper and lower covers, and with joints to substructure integral in the covers. This meant redesigning the V-joint concept developed in the Conceptual Wing Program.

To avoid crossing members for chordwise loads and to preserve the efficient V-joint, the two alternate truss orientations of Figure B-2 were considered. The truss rib concept oriented the truss corrugation in a vertical plane, like conventional ribs, though they were interconnected. The truss spar concept oriented the truss corrugation in a chordwise fashion to provide cover support, like ribs, and reaction paths for chordwise loads. It also offered the attractive possibility of removing front and rear shear webs in a non-fuel-carrying application, such as a horizontal stabilizer, since the truss web edges in conjunction with cover edges formed trusses to carry vertical shear.

The V-joint was basic to the truss web concept. Simple in concept, it featured unidirectional fibers wrapped around a bolt-bearing block or, in the case of fillet bonding, only a thick adhesive bondline between wraparound fibers (truss web) and the adherend (wing cover). It was anticipated most of the major chordwise loads would be introduced from outside the box through the truss crests (except where truss crests do not fall in the right places). This can be done in the cases where external loads are introduced through external plates, such as pylon horizontal reactions and flap lower-surface attachment reactions. The other flap reactions require introduction through the rear web, and load paths must be provided which are not conveniently present in the original truss web arrangement. Fuselage/wing vertical reactions are also taken more efficiently at the fuselage side into trapezoidal shearpanels as in the metal baseline airplane design rather than by connecting directly to truss crests. These changing major joint requirements motivated the investigation of alternate truss geometries (Figure B-2).

Since the alternate truss concepts suffered the same low fabrication cost and feasibility ratings as most of the basic truss web versions, it was decided not to pursue their design pecularities further in this program. The most practical and potentially lowest-cost truss web was retained for tradeoff with the more conventional multirib concept. (See Paragraph 2.4.2, Wing Box Concept.)

The foregoing truss web box concept investigation was accomplished as part of a Douglas Independent Research and Development (IRAD) program (Reference 8).

B-3 MULTIRIB CONCEPTS

It was felt by some, during the preliminary concept evaluation process, that internal load distribution would likely not rearrange itself after 30 years of structural optimization, that there is a close and direct similarity between good design detailing from an analysis and manufacturing standpoint, and that simple, direct load paths are stronger, lighter, and easier to fabricate reliably. Therefore, even though these same design principles could be applied to truss web, it was felt that the multirib, rather than any of the truss concepts, had the greatest potential for low-cost, high-reliability composite wing/empennage design for this program. The potential benefits of composites in this application lie in the elimination of detail parts through fabrication of large integral assemblies. The practical truss concepts appeared to be acquiring more and more parts and complexity; therefore, direct cost estimates for competing designs were obtained. (See Paragraph 2.4.2, Wing Box Concept.)

APPENDIX C
MANUFACTURING COST ESTIMATING DRAWINGS

This Appendix contains the detailed MCE drawings used to develop program manufacturing costs, and the rationale for their development. The drawings were used throughout the program to transmit basic information required for cost evaluation, strength analysis, and weight analysis.

The drawings were developed by an interaction of design and manufacturing personnel to contain structural details which have an impact on manufacturing costs. In order to develop as many areas as possible within program scope, only cost-important areas were considered in detail. Thus, some detail design features are only representative of cost and weight values and are not the result of detailed analysis or trade studies.

STUDY MCE DRAWINGS

Figures C-1 through C-5 present the final drawings used in this study. The five drawings were selected as the minimum required to adequately describe the composite aircraft. The drawings are as follows:

- Wing Structural Box, Figure C-1

  This drawing shows the complete wing box and details the composite face sheet layup for skin panels and spars. Major details are shown.

- Fuselage, Figure C-2

  The complete fuselage is shown on this drawing together with details of structure and splices. Frame and panel composite requirements are detailed. Vertical tail/fuselage and landing gear to fuselage interfaces are shown.

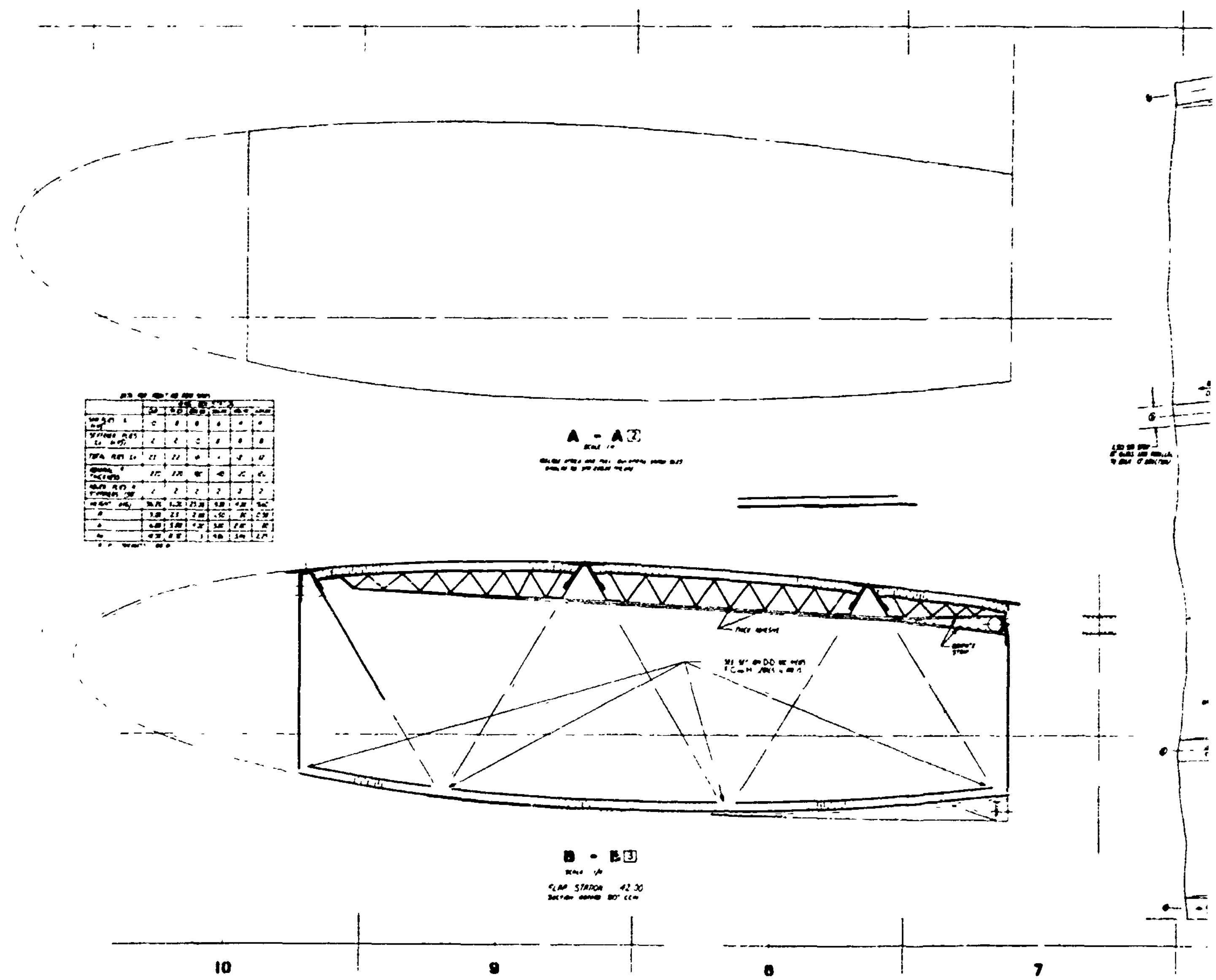
- Empennage, Figures C-3 and C-4

  The basic empennage structure is shown on these drawings which detail the skin panels and substructure for the vertical and horizontal stabilizers. Structure for the movable surfaces is also illustrated on the vertical stabilizer drawing. The rudders are typical of the elevators, spoilers, and ailerons.

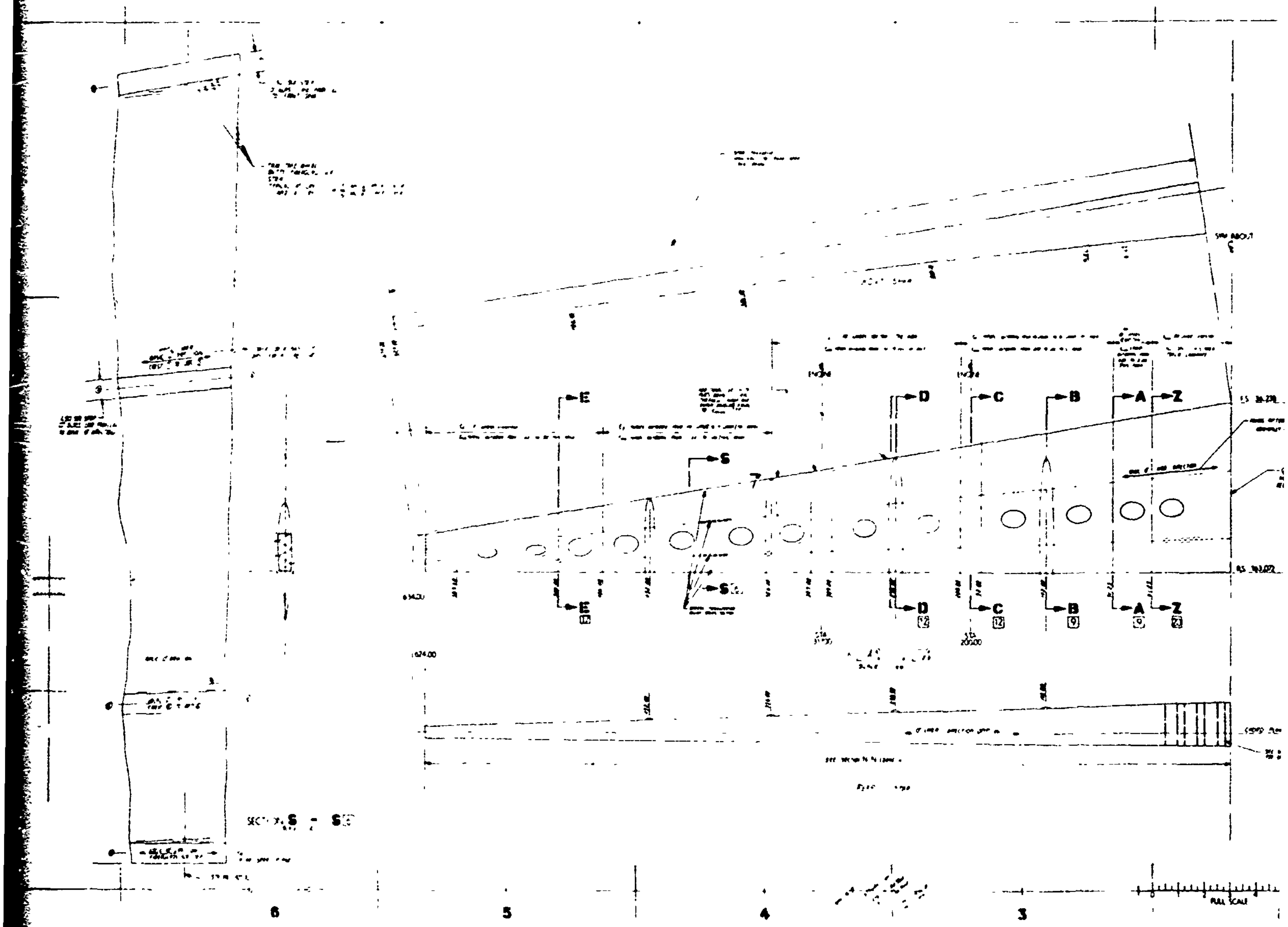
- Structural Scope, Figure C-5

  This drawing illustrates the general structural arrangement and major interfaces, such as the wing/fuselage intersection, flap/wing box drive station, pylon/wing box intersection, etc.

An estimating factor for undefined primary structure will be noted in the general notes of the above MCE drawings. This is a recognition of the necessarily incomplete nature of the design effort. The factor was not used as a multiplier by the estimating groups but as a guide only to indicate major areas for which it was not possible to suggest concepts within program limitations.


A - A 2
B - B 3
FLAP STATION 42 30
10
9
8
7

[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]



2

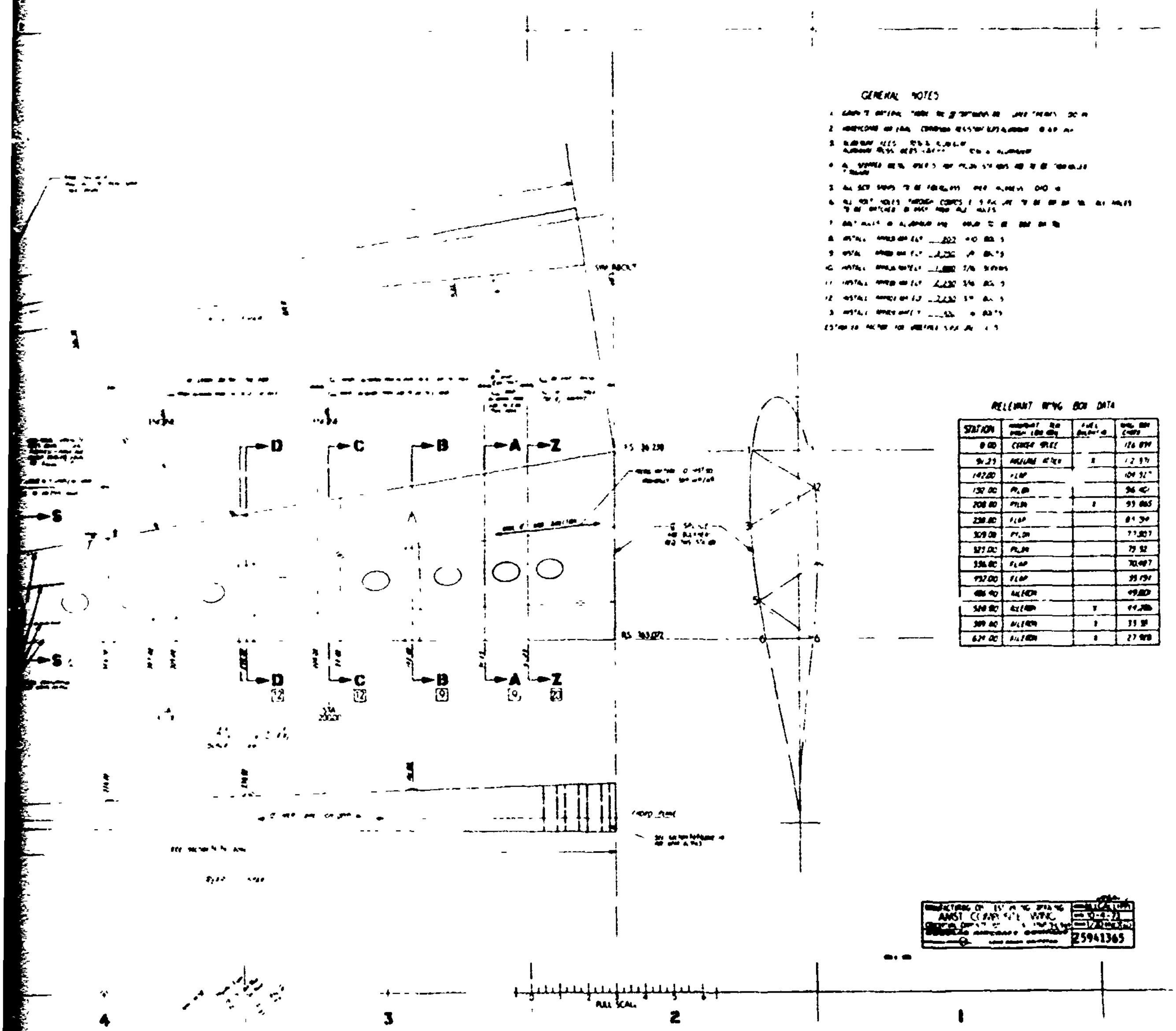
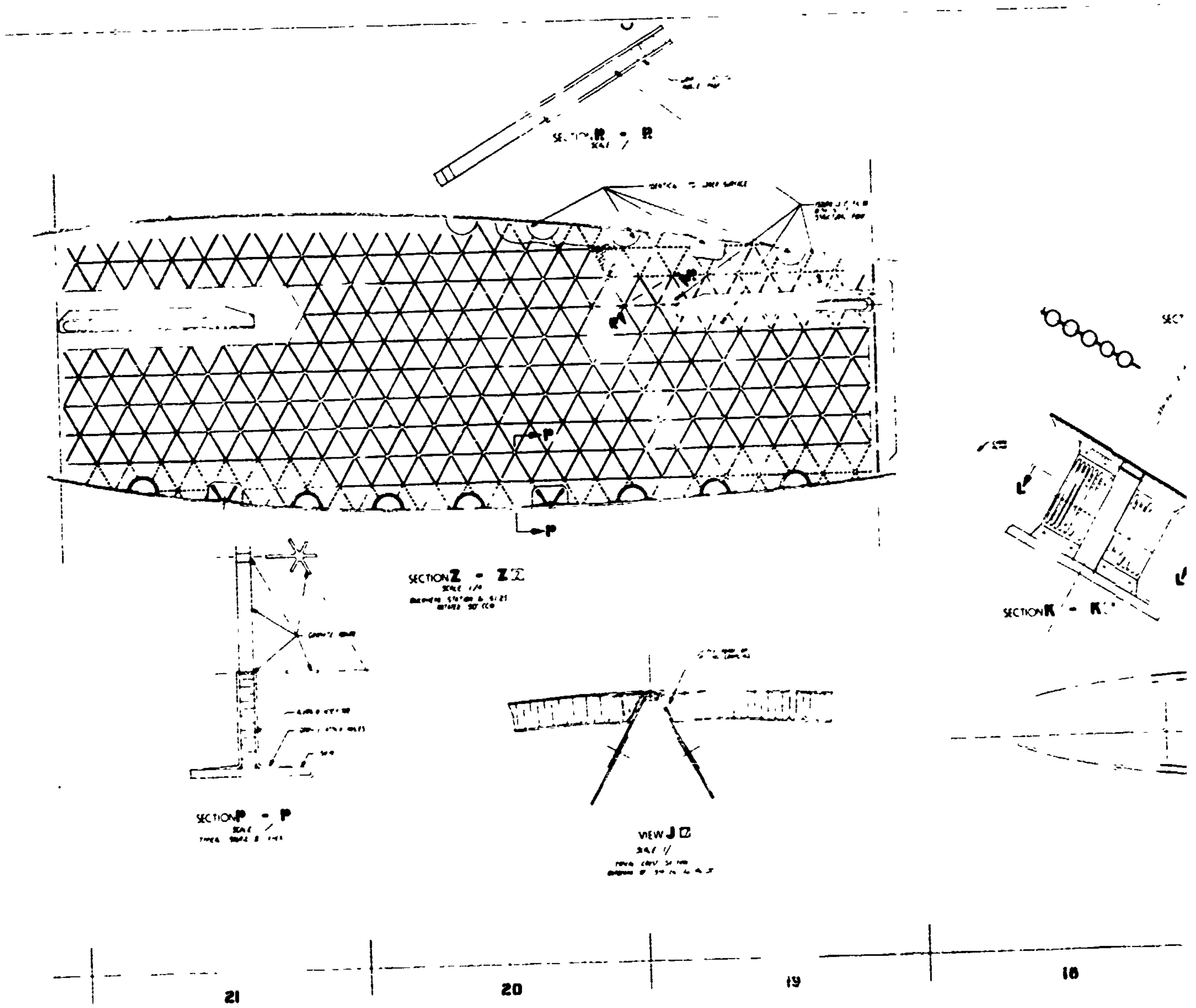
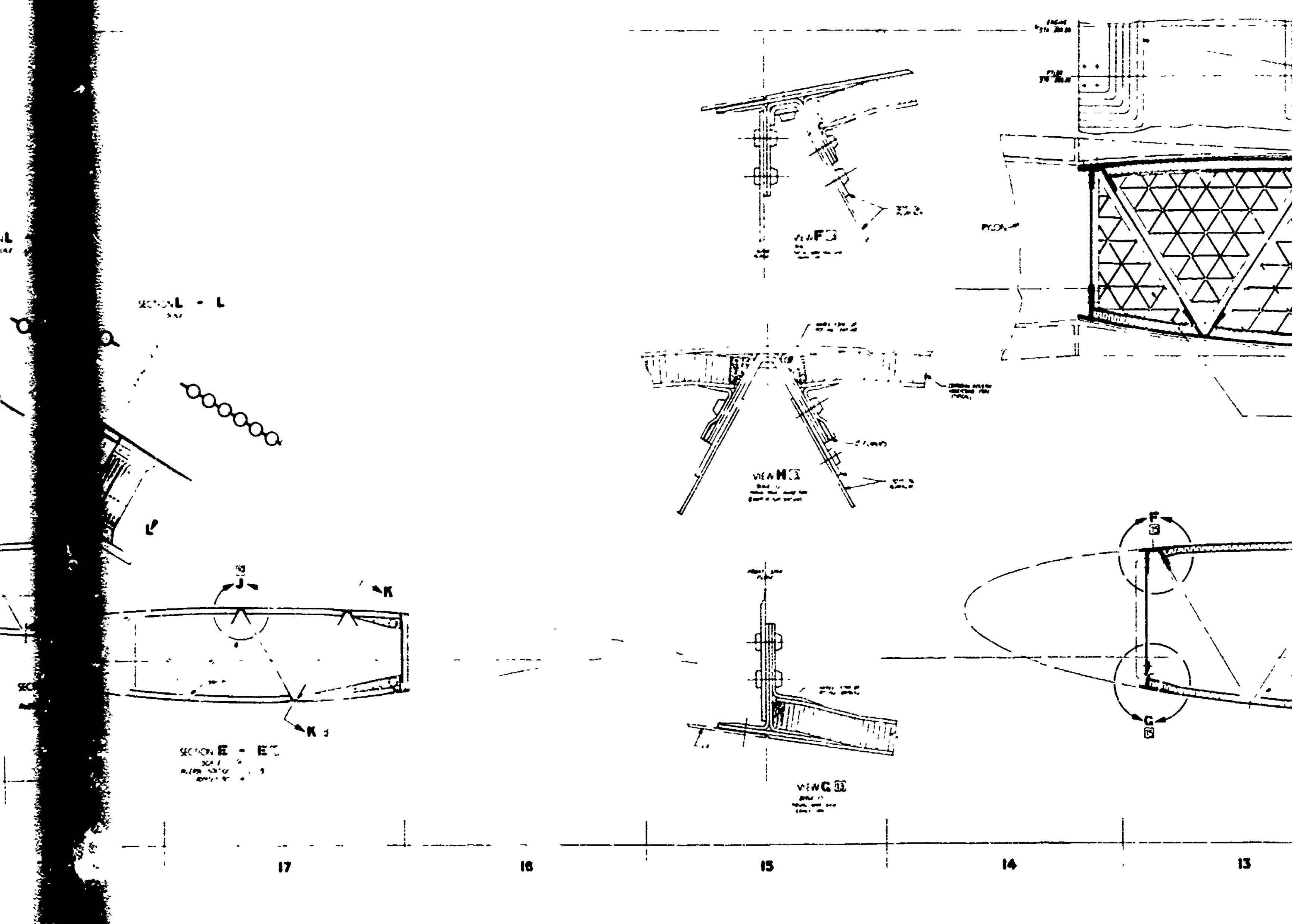


FIGURE C-1. COMPOSITE WING BOX


SECTION R - R
SECTION Z - Z
SECTION P - P
VIEW J
SECTION K - K
21
20
19
18



2

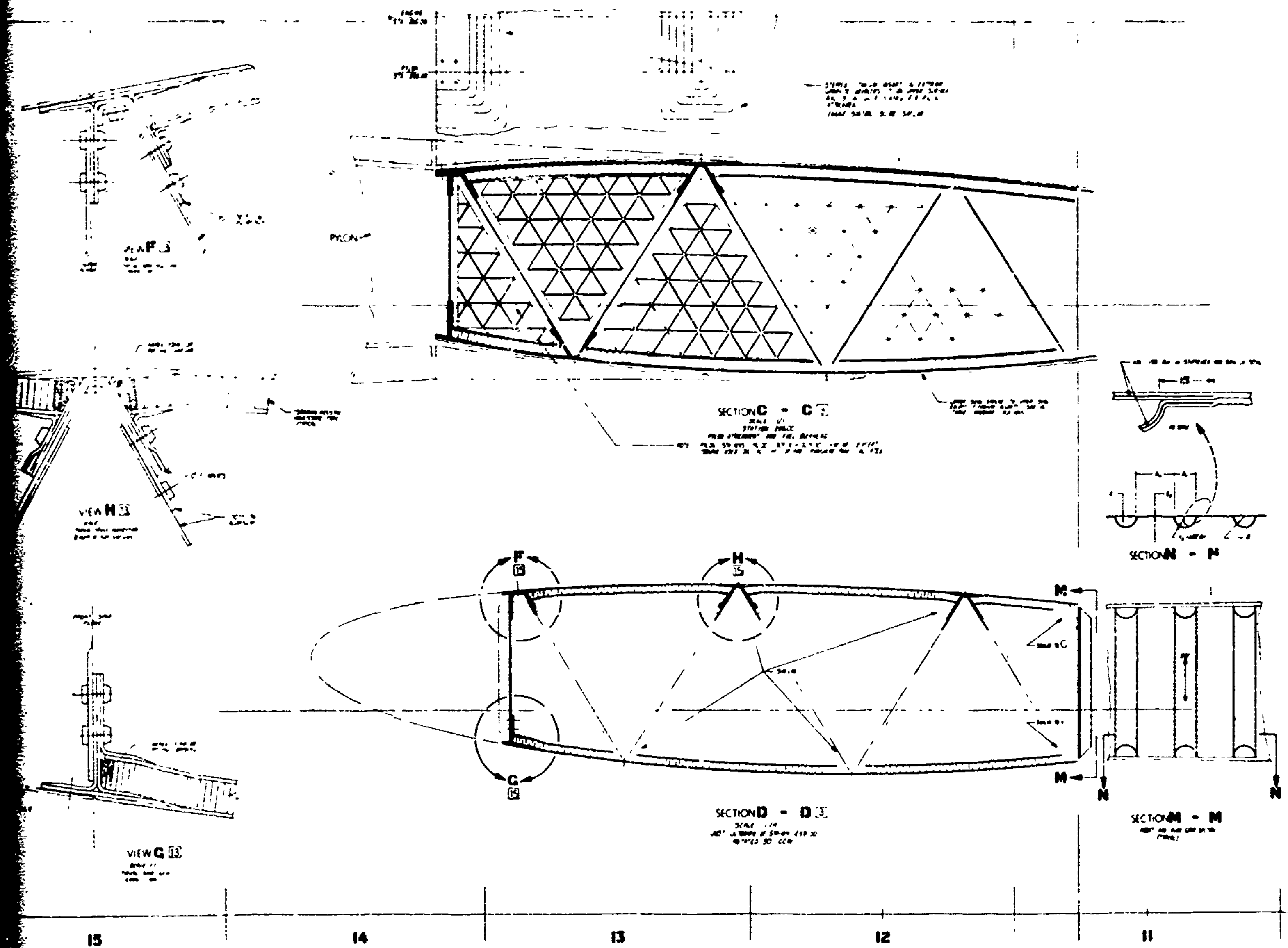
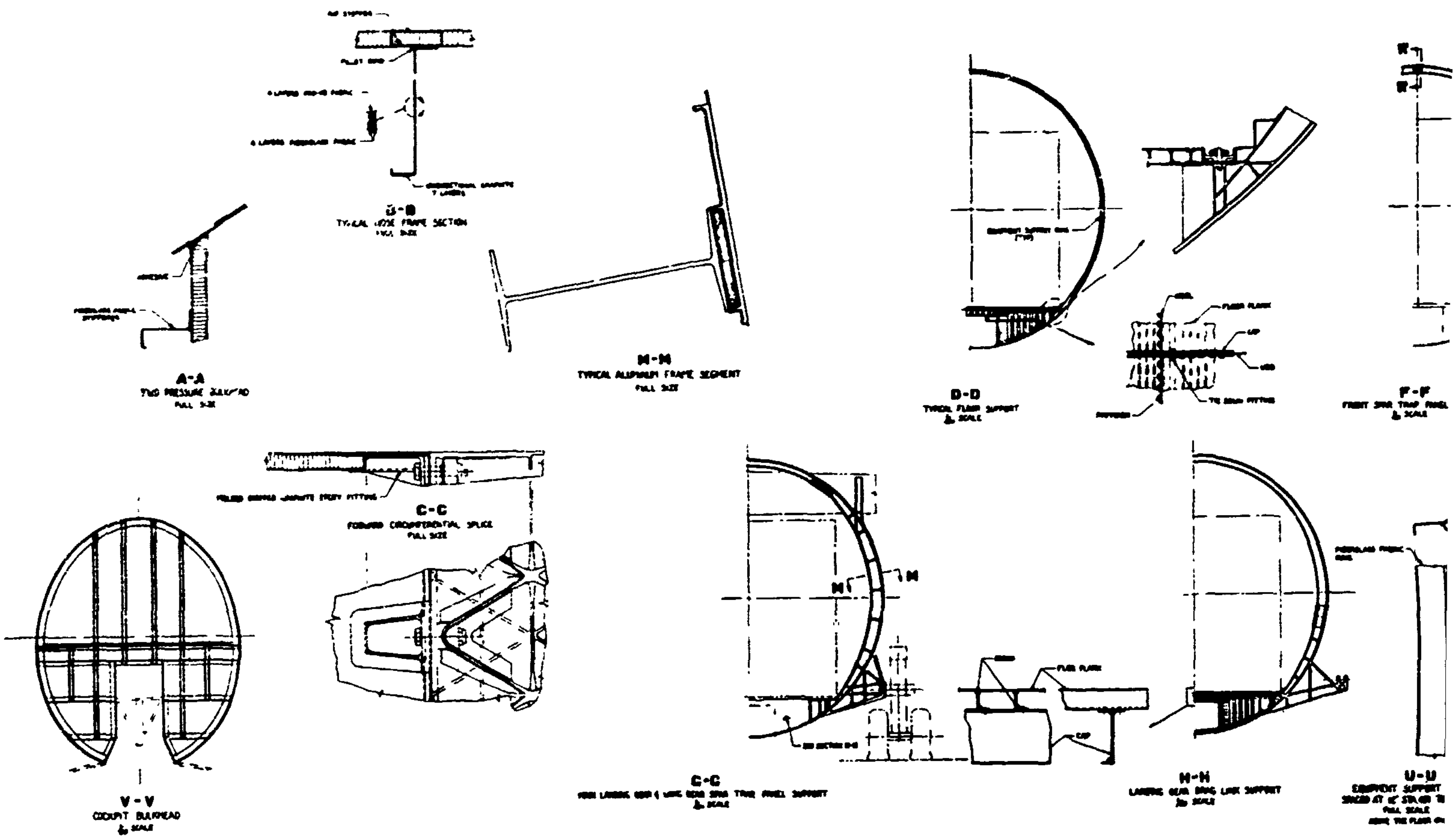
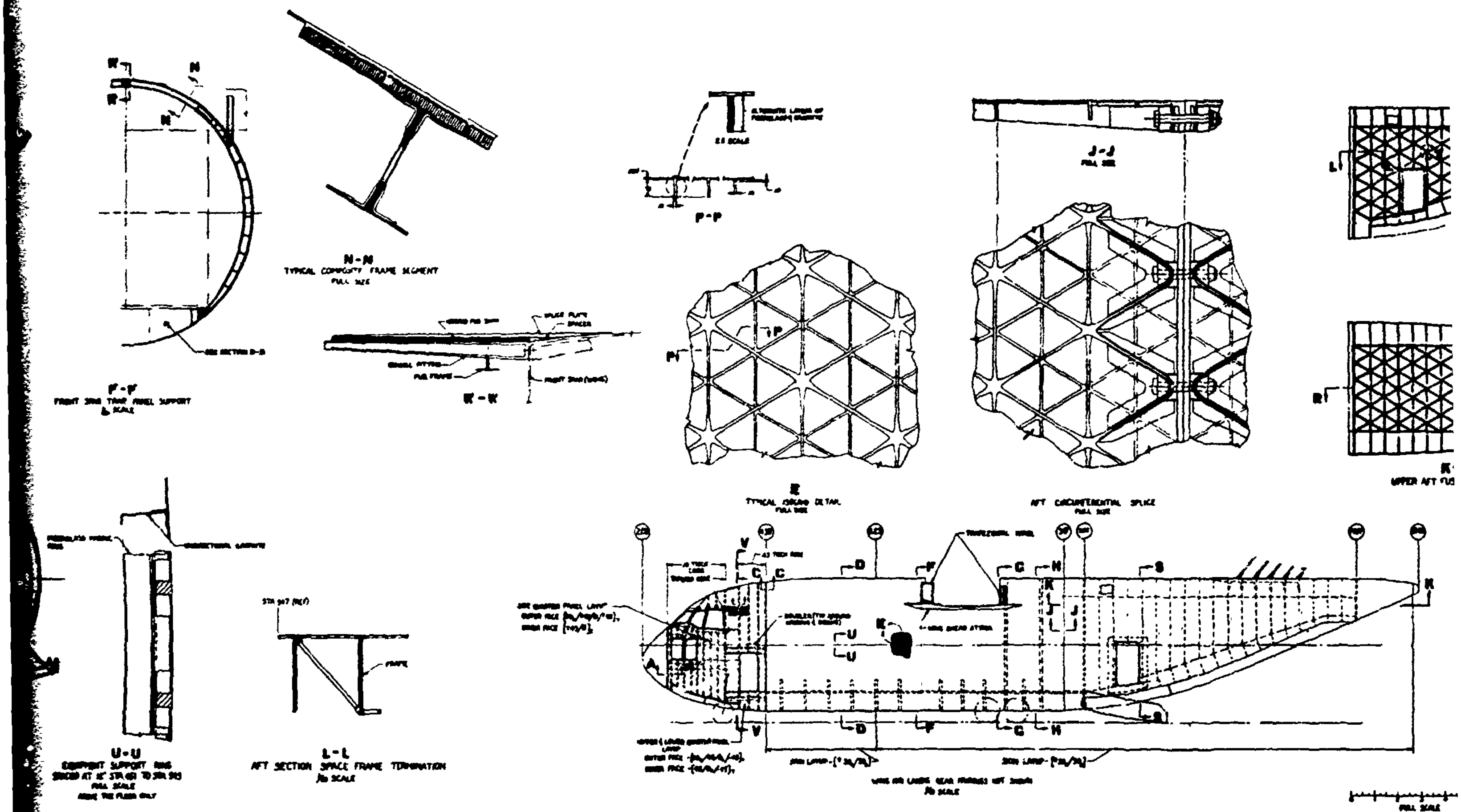


FIGURE C-1. COMPOSITE WING BOX (CONCLUDED)


B-B
TYPICAL NOSE FRAME SECTION
FULL SIZE
A-A
TWO PRESSURE BULKHEAD
FULL SIZE
H-H
TYPICAL ALUMINUM FRAME SEGMENT
FULL SIZE
D-D
TYPICAL FLOOR SUPPORT
F-F
C-C
FORWARD CIRCUMFERENTIAL SPLICE
FULL SIZE
V-V
COCKPIT BULKHEAD
C-C
H-H
U-U
EQUIPMENT SUPPORT
FULL SCALE


N - N
TYPICAL COMPOSITE FRAME SEGMENT
FULL SIZE
F - F
FRONT SPAR TANK PANEL SUPPORT
K - K
P - P
J - J
FULL SIZE
R
TYPICAL ISOGRID DETAIL
FULL SIZE
AFT CIRCUMFERENTIAL SPLICE
FULL SIZE
UPPER AFT FUS
U - U
EQUIPMENT SUPPORT RING
L - L
AFT SECTION SPACE FRAME TERMINATION
FULL SCALE

2

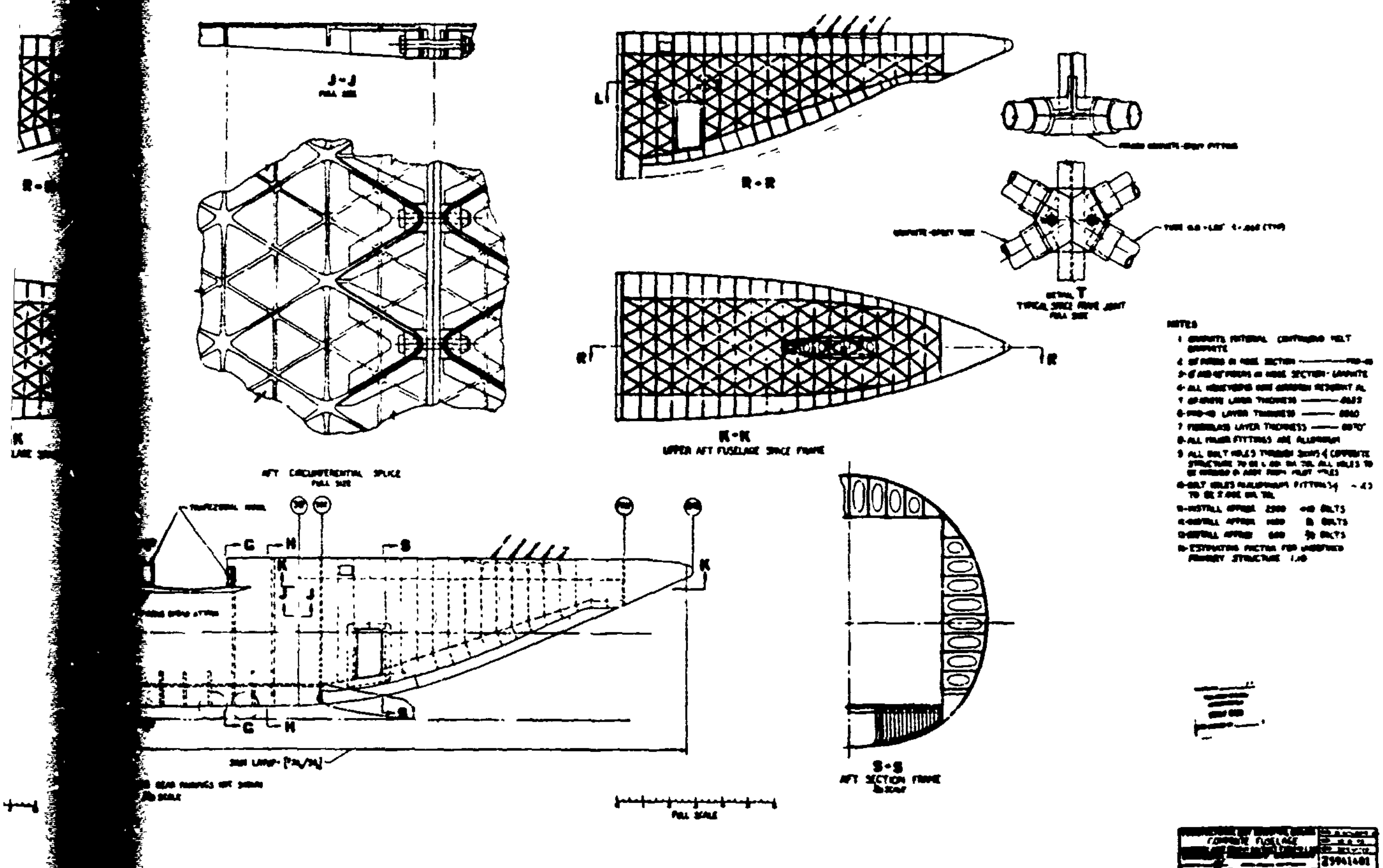
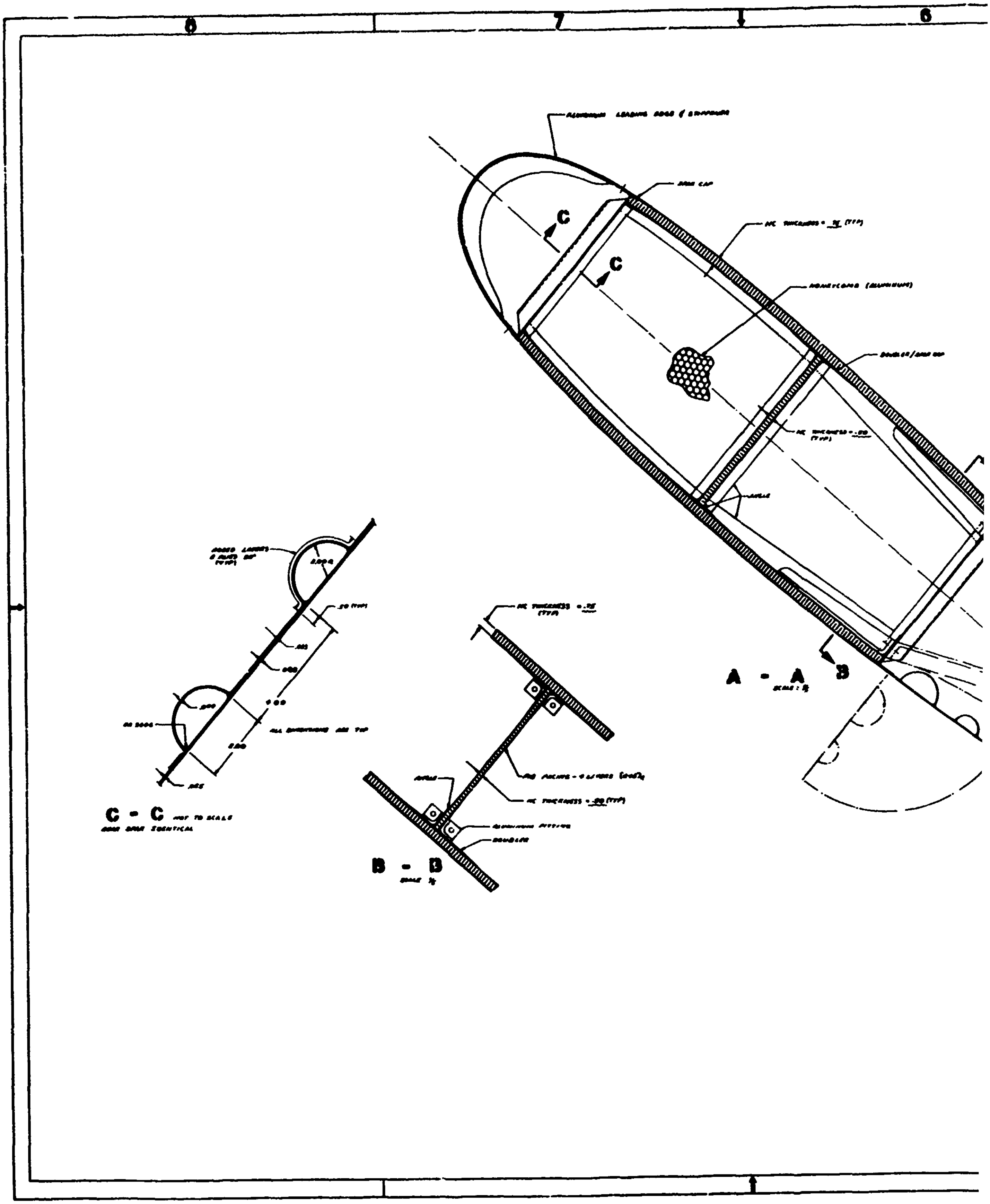
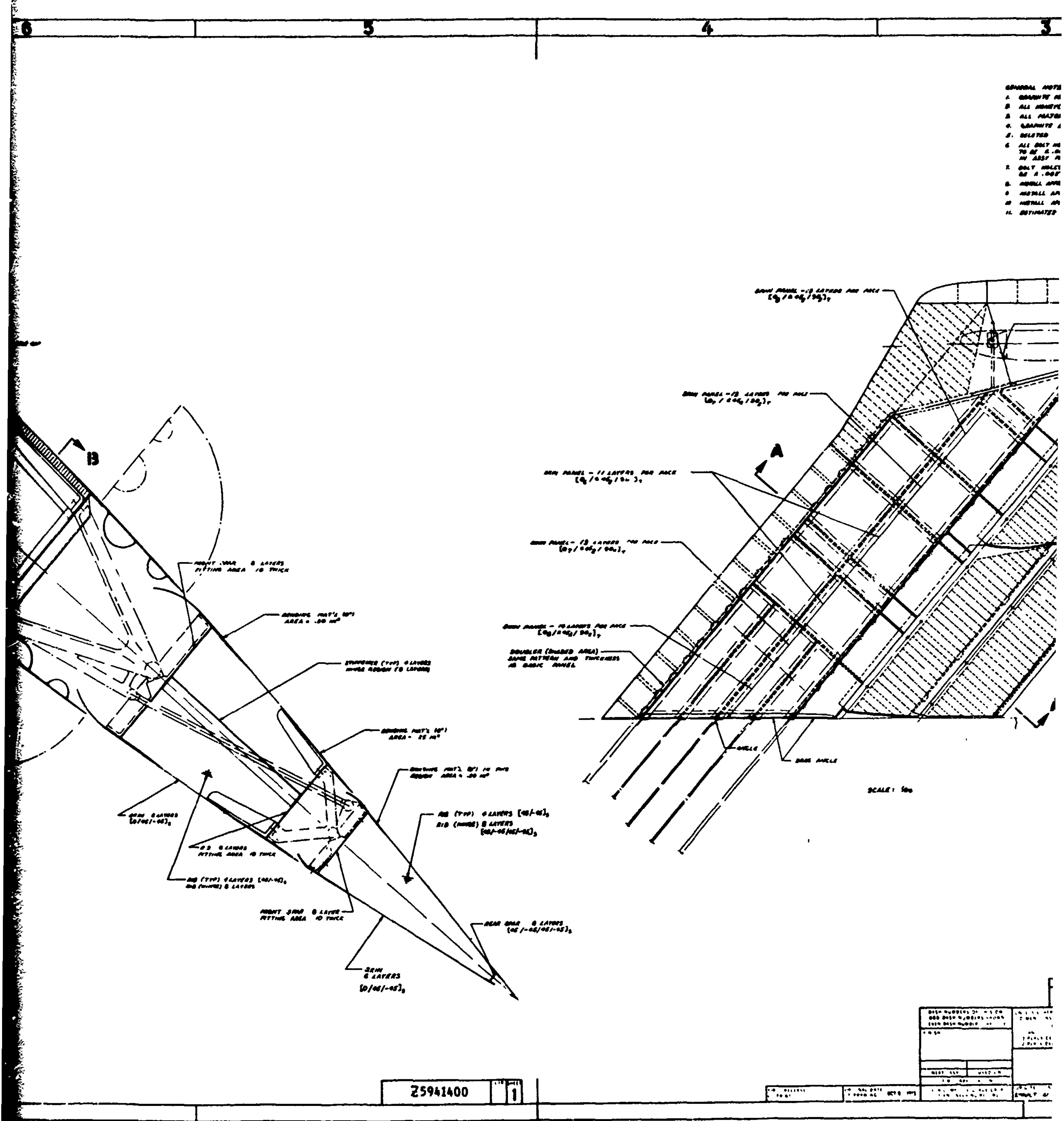


FIGURE C-2. COMPOSITE FUSELAGE


8
7
6
C
C
A - A
B
B - B
C - C


6
5
4
3
B
A
Z5941400
1

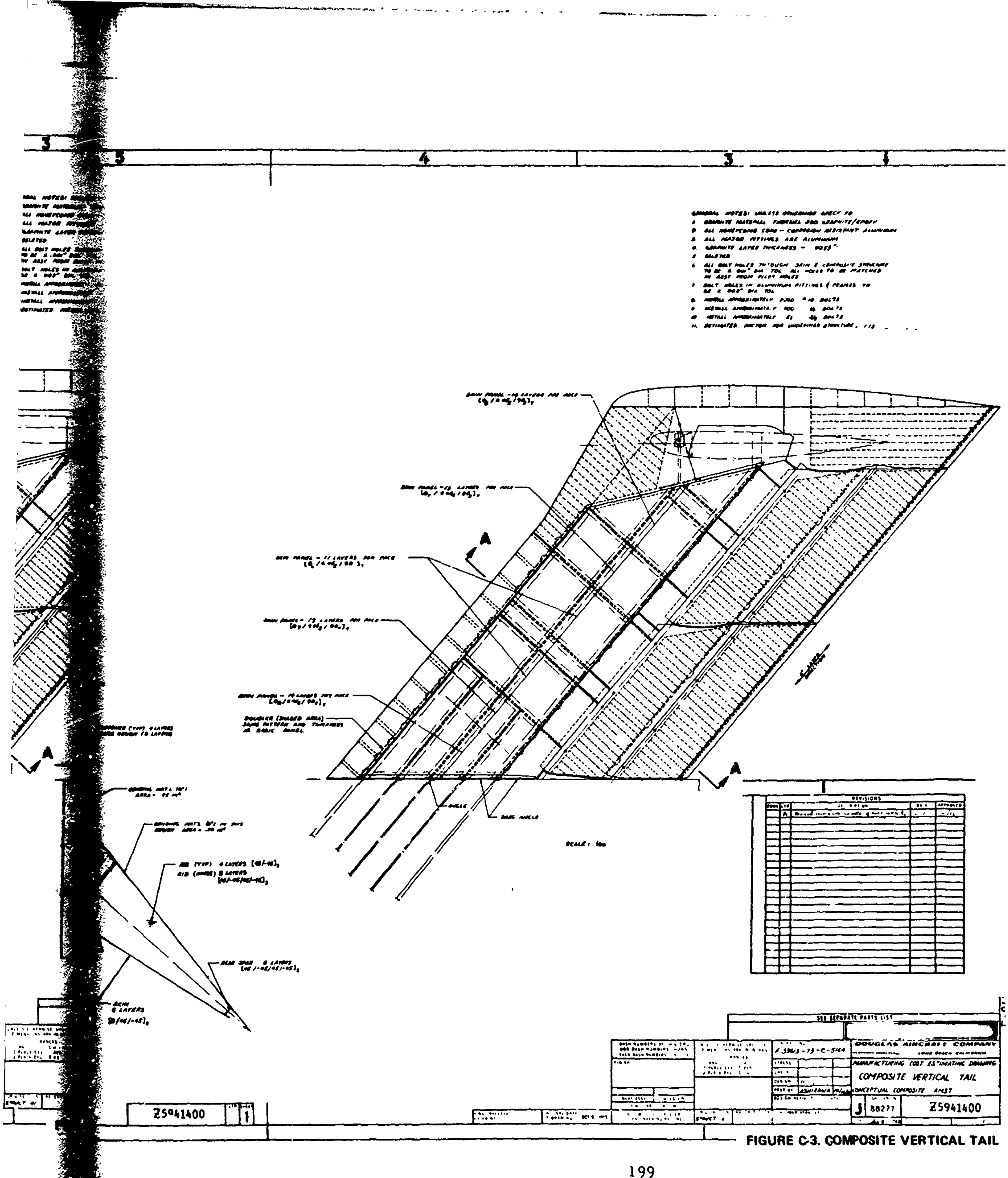
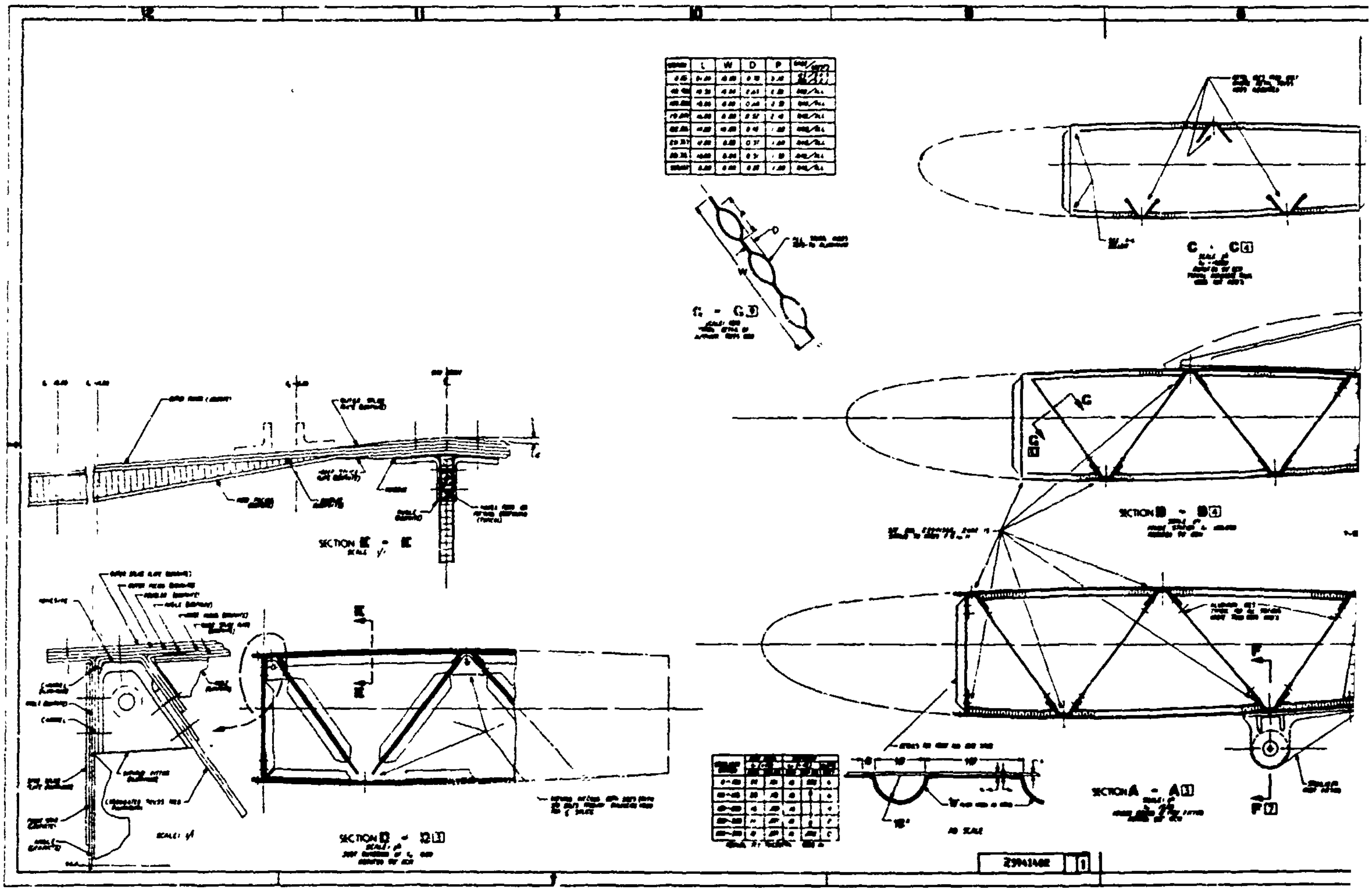


FIGURE C-3. COMPOSITE VERTICAL TAIL



SECTION E - E
SECTION D - D
C - C
G - G
SECTION B - B
SECTION A - A

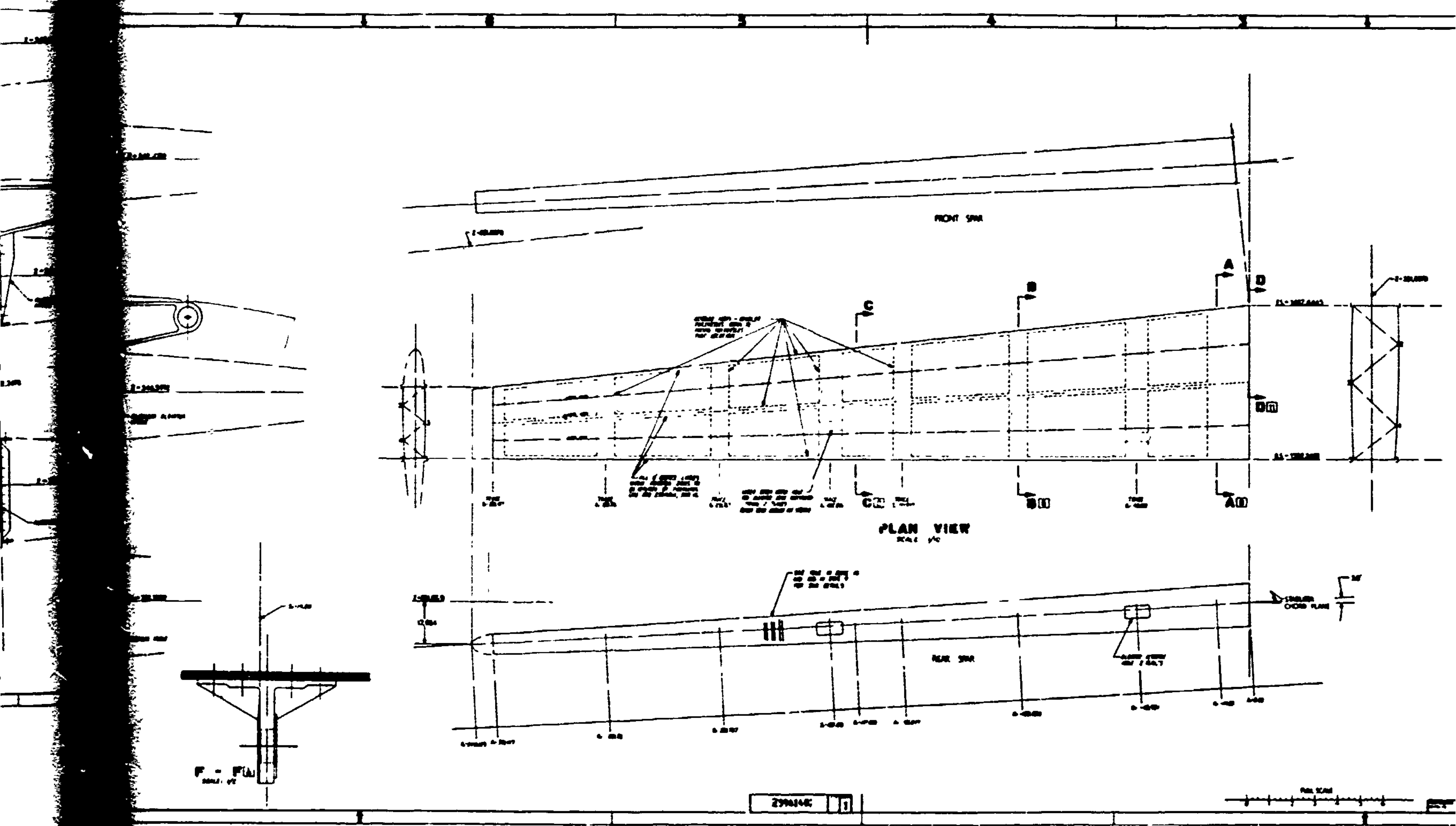


FIGURE C-4.

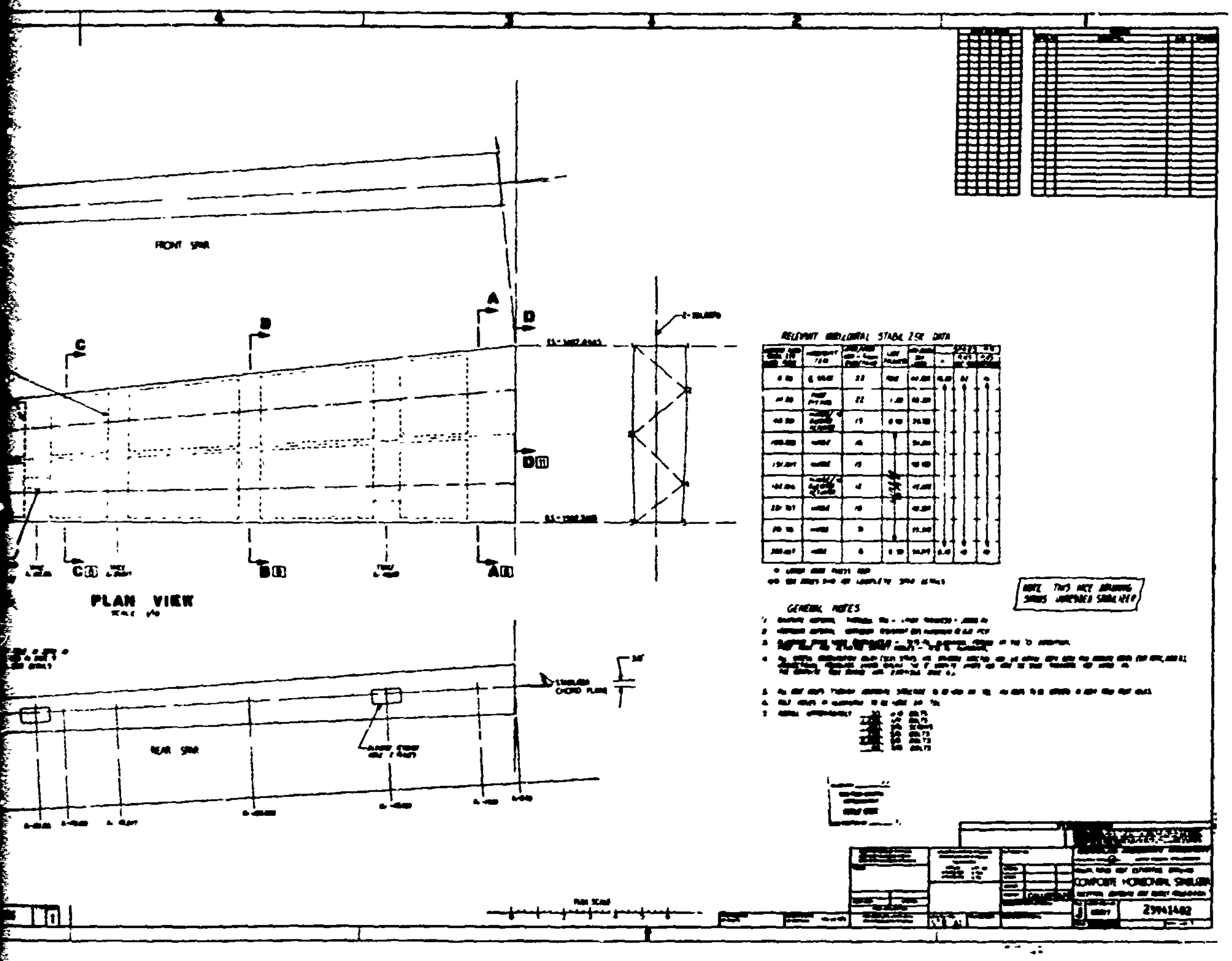
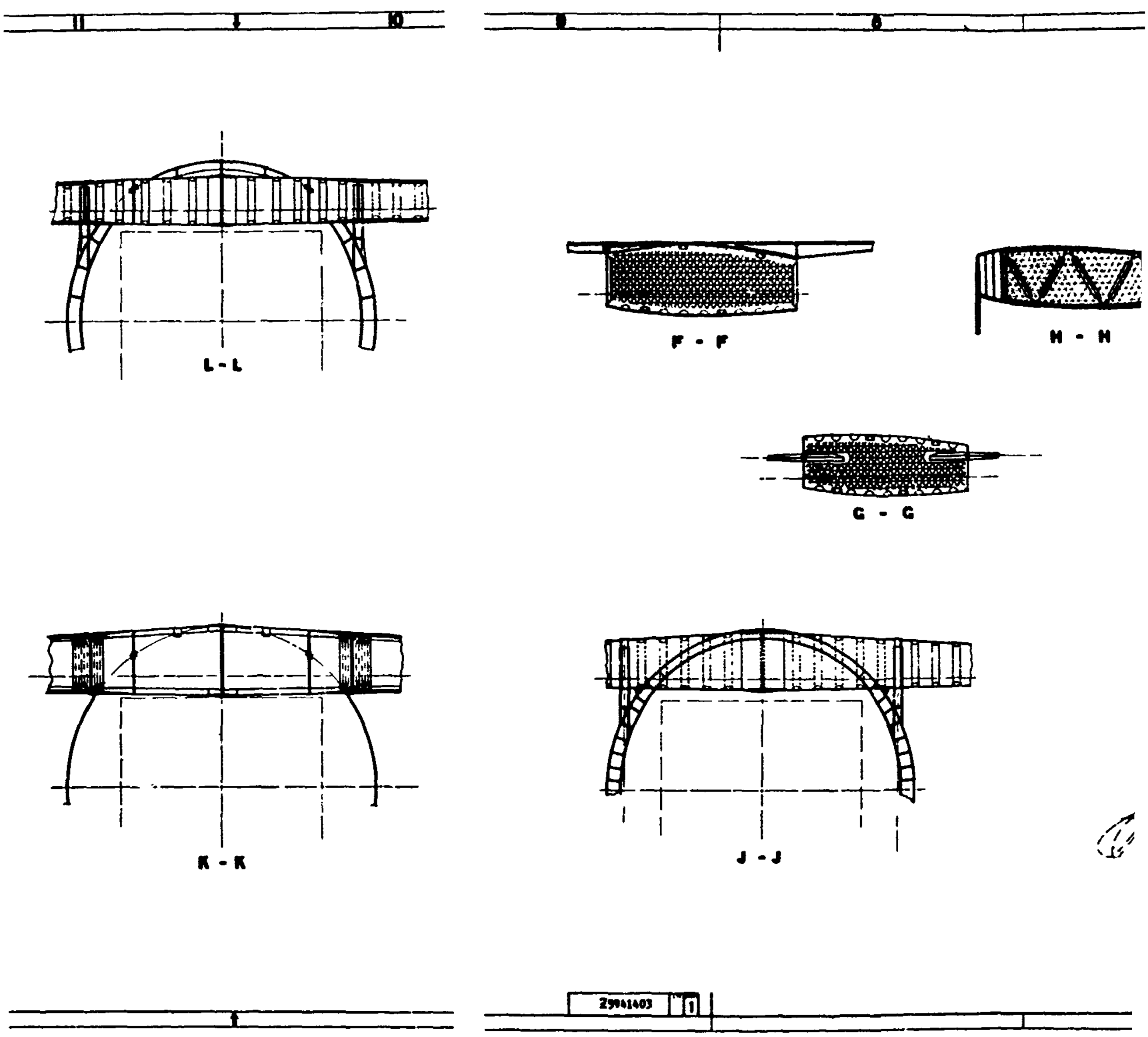
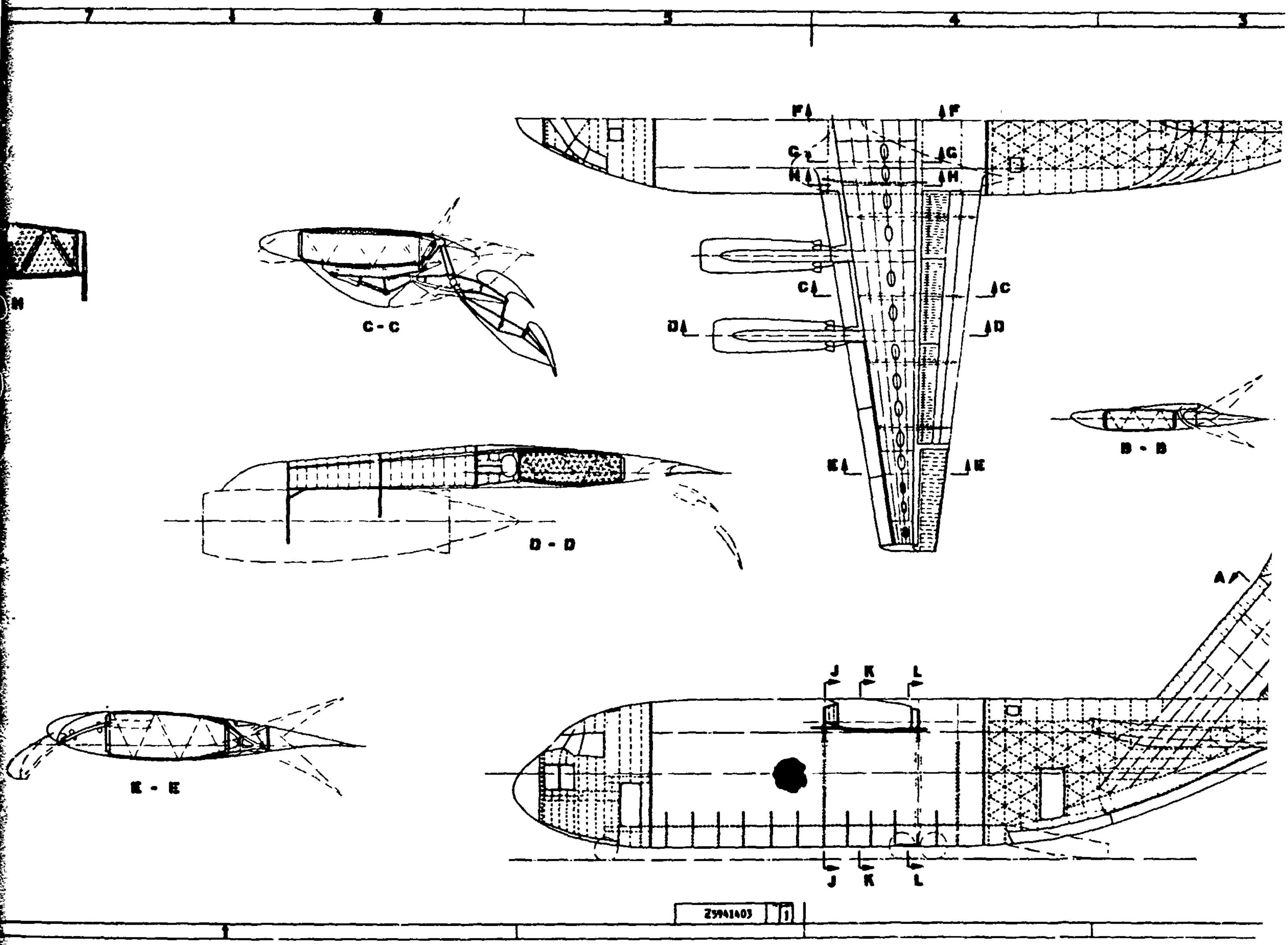


FIGURE C-4. COMPOSITE HORIZONTAL STABILIZER


11
10
9
8
L - L
F - F
H - H
G - G
K - K
J - J
29941403
1



2

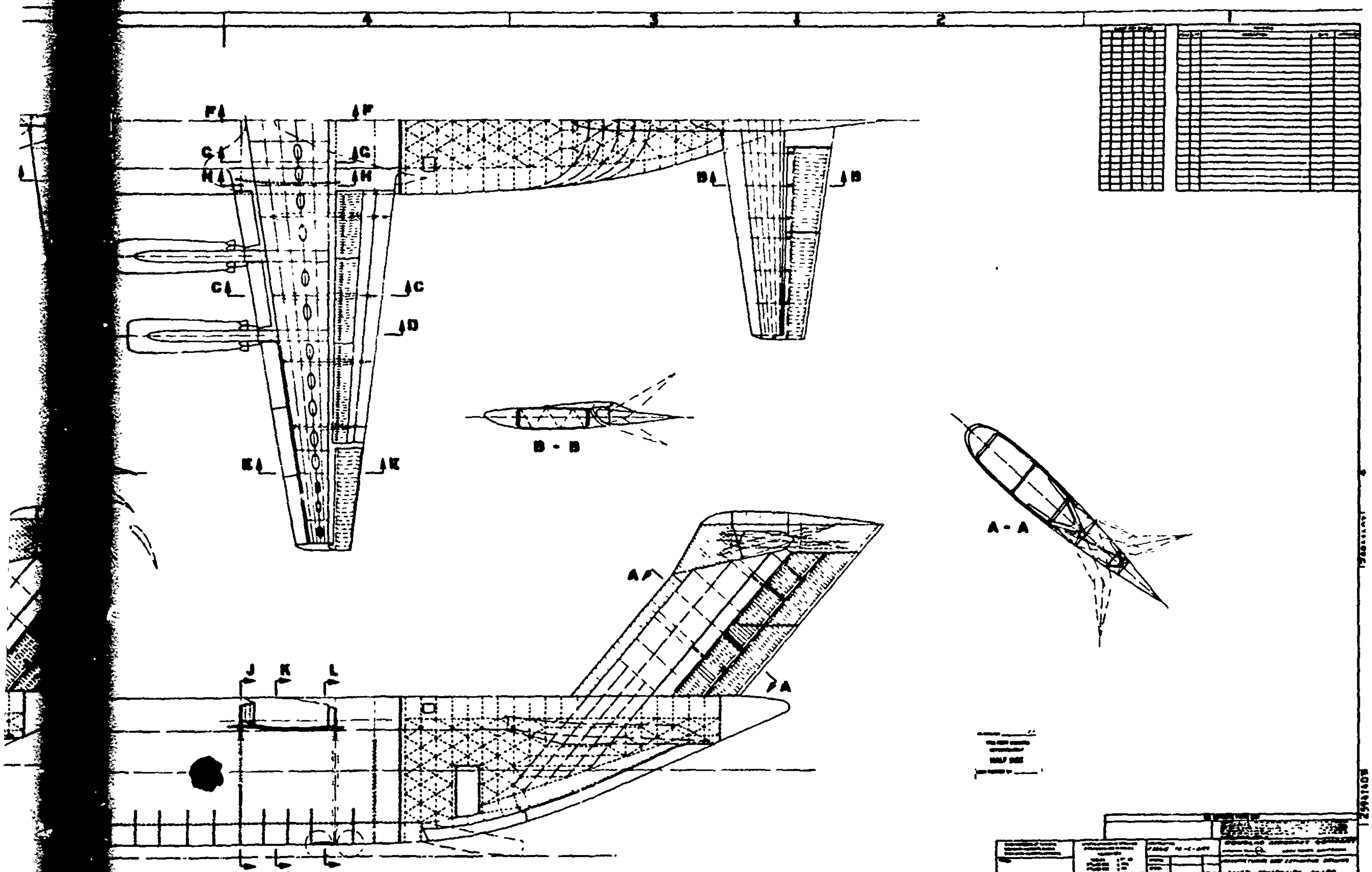


FIGURE C-5. COMPOSITE AMST SCOPE

In order to validate the content and concept of the MCE drawings, the drawing development was made based on a Douglas program for which detailed cost data are available, Reference 4. The resulting drawing, Figure C-6, was developed from the production drawing of the same part. Several iterations of level of detail and cost estimation compared to the actual costs evolved the following set of minimum items to be contained in the drawing:

1. General plan and details of major subassemblies.

2. Sketches to detail layup in nontypical areas; buildups and substructure, for example.

3. Specification of metal parts.

4. Fastener callouts with estimates of number of each type.

5. Notes to explain miscellaneous details if they are not shown; metal shims, and clips, for example.

6. Specification of subassembly interfaces; tolerances, whether machined or not.

7. Specification of all unusual tolerances, particularly for attachments.

MCE drawings were first used in the program discussed in Reference 2.

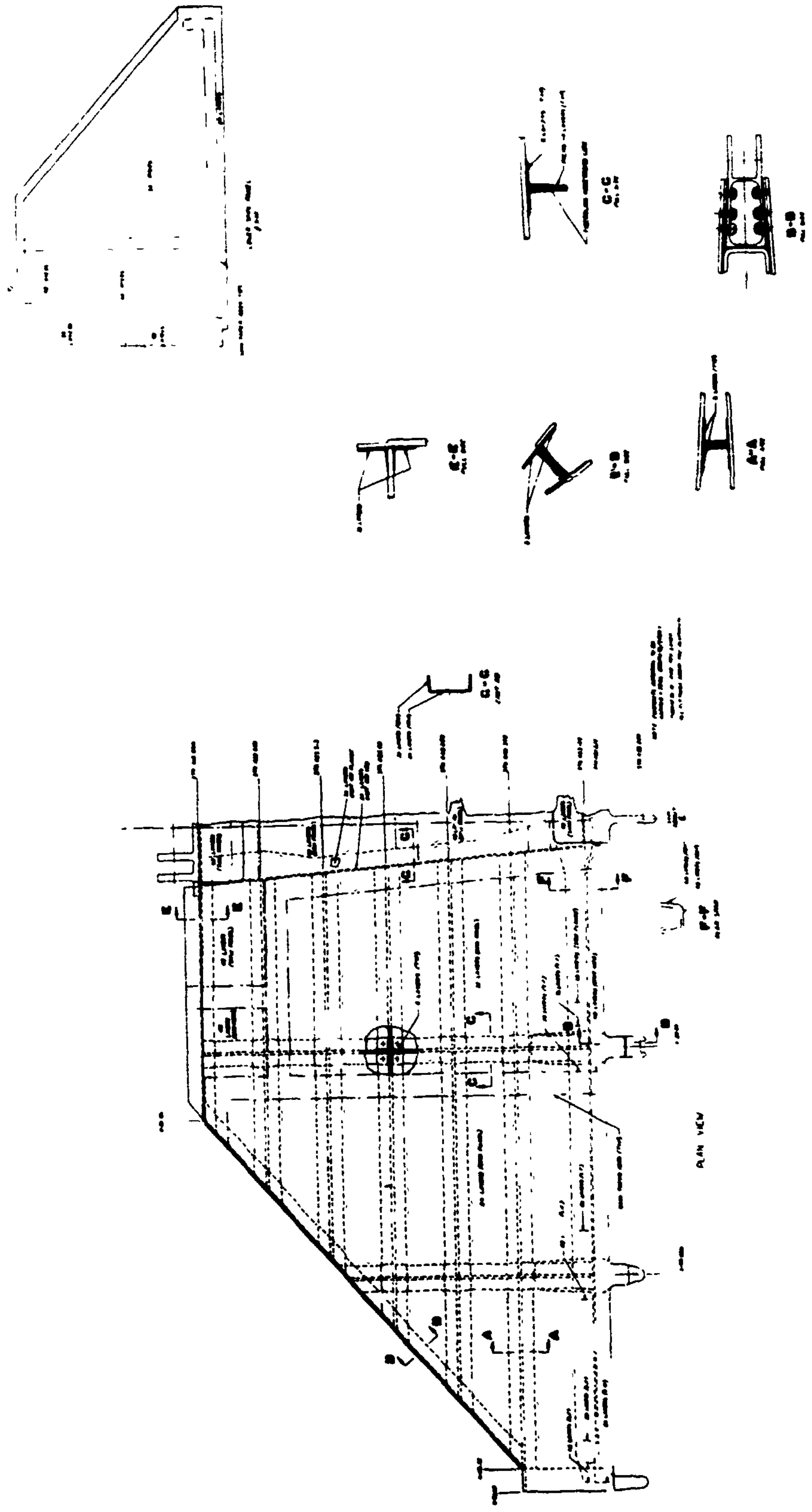


FIGURE C-6. A-4 STABILIZER MCE VALIDATION DRAWING

☆U.S.Government Printing Office: 1975 — 657-019/446